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CONFIGURATION OPTIMIZATION OF A CLASS OF HYPERSONIC CRUISE VEHICLES

RICHARD C. WALKER, MAJOR, USAF
HYPERSONIC RESEARCH LABORATORY

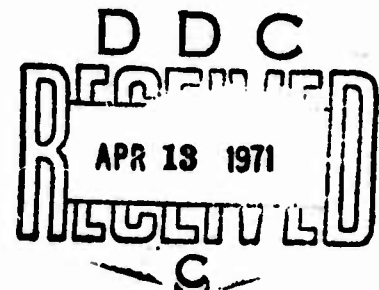
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AIR FORCE SYSTEMS COMMAND

United States Air Force



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**AEROSPACE RESEARCH LABORATORIES
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE
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Foreword

The author would like to acknowledge the guidance of Dr. W. L. Hankey (ARL) throughout the course of this research. This report was prepared by the Hypersonic Research Laboratory, Aerospace Research Laboratories, Air Force Systems Command, United States Air Force, under Project 7064, entitled "High Velocity Fluid Mechanics." This work was also submitted in partial fulfillment of the requirements for the degree of Doctor of Philosophy at the Air Force Institute of Technology.

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Nomenclature

Symbol

| | |
|-----------|--|
| A | area |
| AR | wing aspect ratio |
| b | fuselage width |
| b_w | exposed wing span at midcord |
| c | longitudinal length of wing |
| c_p | pressure coefficient, $c_p = \frac{p - p_1}{q}$ |
| C_D | drag coefficient, $C_D = \frac{D}{q \ell_b b}$ |
| C_{D_b} | nondimensional fuselage drag, $C_{D_b} = \frac{D_b}{p_1 \ell_1 b}$ |
| C_{D_w} | nondimensional wing drag, $C_{D_w} = \frac{D_w}{p_1 \ell_1 b}$ |
| C_f | local skin friction coefficient |
| C_{L_b} | nondimensional fuselage lift, $C_{L_b} = \frac{L_b}{p_1 \ell_1 b}$ |
| C_{L_w} | nondimensional wing lift, $C_{L_w} = \frac{L_w}{p_1 \ell_1 b}$ |
| C_T | nozzle force coefficient, $C_T = \frac{F_n}{p_1 \ell_1 b}$ |
| D | drag |
| D_b | fuselage drag |
| D_w | wing drag |
| F_n | resultant nozzle force |
| f | fuel-air ratio |
| g_c | conversion constant |
| H | altitude; enthalpy |
| h | fuselage height at station 5 (combustor exit) |
| h_4 | vertical distance of cowl from under surface of fuselage |

| | |
|-----------------------------|--|
| h_b | fuselage base height |
| I_{sp} | specific impulse |
| i | initial |
| $\hat{i}, \hat{j}, \hat{k}$ | unit vector in x, y, z direction respectively |
| L | lift |
| L_b | fuselage lift |
| L_n | nozzle-aftfuselage lift |
| L_o | length of ray in design space |
| l | longitudinal dimension on fuselage |
| l_b | longitudinal length of the fuselage |
| l_{ew} | longitudinal length from the nose of the vehicle to the leading edge of the wing |
| l_1 | scale factor; longitudinal length of first inlet ramp |
| M | Mach number |
| m | number of constraint relations |
| N | number quantities defining model |
| n | number of degrees of freedom in model |
| \hat{n} | vehicle surface outward normal unit vector |
| nm | nautical mile |
| p | static pressure |
| \dot{Q}_c | heat absorbing rate of the regenerative coolant |
| \dot{Q}_h | convective heat transfer rate to a surface |
| \dot{Q}_{le} | convective heat transfer rate to a leading edge |
| q | dynamic pressure, $q = \frac{\gamma}{2} p M^2$ |

| | |
|--------------------------|---|
| R | cruise range; gas constant |
| RF | Breguet range factor, $RF = \frac{L}{D} I_{sp} \frac{V_1}{1 - V_1^2 / V_o^2}$ |
| r_e | earth radius |
| r_n | leading edge radius |
| \bar{S} | structural parameter set |
| S_t | Stanton number |
| S_p | exposed wing planform area |
| T | static temperature; thrust |
| SF | Breguet structural factor, $SF = \ln \frac{W_i}{W_f}$ |
| t | time |
| \hat{t} | vehicle surface tangent unit vector aligned with the local flow direction |
| V | magnitude of velocity vector |
| Ψ | volume |
| V_o | orbital velocity (26,000 ft/ sec) |
| W | weight |
| \dot{w} | weight rate of flow |
| \bar{X} | design variable set, $\bar{X} = (X_1, X_2, \dots, X_n)$ |
| (X_1, X_2, \dots, X_n) | components in a n-dimensional design variable space |
| x, y, z | rectangular cartesian coordinate system |

Greek Symbols

| | |
|------------|---|
| α | angle of attack |
| γ | ratio of specific heats; flight path angle |
| γ_s | vehicle surface area-structural weight proportionality constant |

| | |
|----------------|--|
| δ | vehicle surface orientation angle |
| ζ | heat absorbing capacity of coolant fuel per pound mass |
| θ | shock wave angle |
| θ_T | resultant nozzle force direction angle measured from a plane normal to the flight path |
| ρ | density |
| σ | ambient-sea level density ratio |
| τ | shear stress |
| ϕ | equivalence ratio, $\phi = \frac{f}{f_s}$ |
| ψ | range angle |
| $\Delta\omega$ | grid size parameter in method of characteristics analysis |

Subscripts

| | |
|-----|---|
| a | air |
| av | average |
| abw | adiabatic wall condition |
| b | body; base |
| c | calculated; cooling; capture; combustor |
| e | equipment; station on nozzle-aftfuselage surface at termination of fuselage |
| f | final; friction |
| fc | cruise fuel |
| fl | total available fuel |
| i | initial |
| max | maximum |
| o | nominal |

| | |
|-------------|---|
| opt | optimum |
| p | payload; propulsive stream; pressure |
| s | structure; stoichiometric |
| st | stationary |
| stag | stagnation point |
| to | takeoff |
| w | wetted; wing; wind; wall |
| 1 | free stream properties; first inlet ramp |
| 2 | properties behind first inlet shock; second inlet ramp |
| 3 | properties behind second inlet shock; inlet cowl |
| 4 | properties behind third inlet shock; station at combustor entrance |
| 5 | station at combustor exit; nozzle-aftfuselage surface |
| 6 | nozzle cowl |
| 7 | upper surface of fuselage |
| 8 | vehicle base |
| 9 | upper surface of wing |
| 11 | lower surface of wing |
| 12 | side surface of wing |

Abstract

An optimization of air breathing hypersonic cruise vehicles was performed in order to determine basic configuration characteristics and performance trends. A distinctive feature of the investigation is that prediction techniques such as the method of characteristics were used to determine the flow field surrounding the vehicle; therefore, any interaction between the aerodynamic and propulsive flow fields is accounted for in a fundamental manner. The general class of vehicles considered in the study cruise in the Mach 8 - 12 speed regime, utilize a hydrogen fueled supersonic combustion ramjet engine, and can be geometrically characterized as two-dimensional wedgelike shapes. Configurations were optimized for maximum cruise range as determined from the Breguet range equation which incorporates a measure of the aerodynamic, propulsive, and volumetric efficiencies of a configuration. A generalized configuration model was defined by discrete parameters which transformed the variational problem to a static or discrete optimization problem. The direct method of function optimization, utilizing search algorithms such as random point and adaptive creeper techniques, was employed to determine the value of the parameters defining the optimum configuration for cruise at design Mach numbers of 8, 10, and 12. The design parameter space in the vicinity of the optimum point was explored to show performance sensitivity. Results of this study showed that optimum cruise configurations are characterized by small wings. Approximately three-fourths of the total lift is provided directly from the propulsive system in accordance with an expression for optimum propulsive force vectoring derived in the study. The results also indicated that Breguet range factors of approximately 10,000 nautical miles are attainable by vehicles which cruise in the Mach 8 to 12 speed regime.

CONFIGURATION OPTIMIZATION OF A CLASS OF HYPERSONIC CRUISE VEHICLES

I. Introduction

Background

The problem considered in this dissertation is the configuration optimization of air breathing hypersonic cruise vehicles. This study is unique in that vehicle shape will be related directly to the flow field surrounding the vehicle. Previous configuration studies relating geometry directly to the flow field have dealt only with components of the vehicle such as the wings (Ref 1). Complete vehicle optimization studies, on the other hand, usually relate the shape of the vehicle indirectly to the flow field through the use of force coefficients and geometric ratios (Ref 2). At hypersonic speeds the interaction between the aerodynamic and propulsive characteristics of the vehicle can become significant, making questionable the use of force coefficients and geometric ratios to relate vehicle shape to the actual flow field surrounding the vehicle.

Kuchemann, writing in "Progress in Aeronautical Sciences" (Ref 3), points out the danger of carrying over concepts derived for components to a system composed of the components. He also indicates the need for a fundamental analysis of systems which includes the combined effect of the aerodynamic and propulsive flow field, which is the overall goal of this dissertation.

Optimization Process

The statement of the overall goal of the optimization study is the first step in the optimization process. According to Pun (Ref 4), the steps involved in the solution of a general optimization problem include: (1) definition of the overall goal, (2) definition of the level of solution, (3) mathematical formulation of the optimization problem, (4) selection of the optimizing method, and (5) realization of results. These steps also serve as a logical outline for the development of the

optimization problem considered in this dissertation.

Level of Solution. In a problem as broad and complex as the optimization of cruise vehicles, the definition of the level of solution becomes very important. For example, the level of solution could range from a cursory feasibility study to a study for implementing a prototype hypersonic cruise vehicle which would be complete in every detail.

The level of solution adopted for this investigation was between the extremes cited in the above example. The investigation can be characterized as a shape study with emphasis placed on relating vehicle geometry directly to the flow field. The level of solution is reflected in the mathematical formulation of the optimization problem which is the next step in the optimization process.

Problem Formulation. In analytical aerodynamic configuration studies, the mathematical formulation of the optimization problem consists of selection of a performance criterion and a model to relate vehicle configuration to the performance criterion. In the literature, the performance criterion is also referred to as a payoff, return, or cost function. Chapter II contains the development leading to the selection of cruise range as the performance criterion for this study.

The ideal vehicle model in an optimization study would be completely general with all variables and factors determined directly from the physics of the problem. In order to make the problem tractable, however, it was necessary to limit the investigation to a particular class of vehicles. Chapter II contains a discussion of the factors involved in the selection of the vehicle model. The selected class of vehicles cruise in the Mach 8 to 12 speed regime, utilize a constant area supersonic combustion ramjet engine which burns hydrogen fuel, and can be characterized geometrically as two-dimensional wedgelike wing body vehicles.

Chapter II also contains a description of the force and flow field prediction methods which are essential to relating vehicle shape directly to the surrounding flow field. Once the mathematical formulation of the problem is complete, the next step in the optimization process becomes the selection of a method to optimize the performance criterion.

Optimizing Method. The two general optimizing methods used in aerodynamic configuration studies are the indirect and direct methods. Until recently, most configuration optimization studies have utilized indirect methods such as the calculus of variations. With the advent of the digital computer, the direct method has become practical in the solution of optimization problems. Chapter III contains a discussion of the application of the direct method of optimization to the cruise vehicle optimization problem considered in this investigation.

Realization of Results. The last step in the optimization process, the realization of results, is closely related to the definition of the level of solution. The level of solution adopted for this study does not reveal all the possible performance and configuration characteristics of optimized hypersonic cruise vehicles. However, the results and conclusions of this study, contained in Chapter IV and V respectively, do answer a number of fundamental questions concerning the performance and configuration of hypersonic cruise vehicles.

II. Problem Formulation

One of the most difficult steps in the process of optimizing a system such as a complete vehicle configuration is the formulation of the problem. At all times, the objective or overall goal of the investigation must be kept in mind as well as the adopted level of solution. Since the goal of the present investigation was to relate optimum vehicle shape directly to the flow field, emphasis was placed on model selection and flow field prediction techniques which allows the flow field quantities to be determined quite accurately.

The problem formulation consists of selecting a performance criterion and model to relate vehicle configuration to performance. Culminating the problem formulation is an IBM 7094 computer program which evaluates the performance of a generalized configuration defined by various sets of design parameters. A discussion of the selection of performance criterion, model, and design parameters is contained in this section, while details of the computer program formulation as well as a listing of the program are contained in Appendix E.

Performance Criterion

For cruise vehicles the primary objective is either to carry a given payload for the maximum distance, or to carry the maximum payload over a given distance. Thus, either payload or range is a reasonable choice for the performance criterion. Maximum range with a fixed payload was chosen as the measure of performance in this investigation. Total range of a vehicle includes range covered in ascent, cruise, and descent. In order to make the problem tractable and still directly relate vehicle shape to flow field quantities, only the cruise phase of the mission profile was considered. Thus, the optimized configuration represents the optimum cruise configuration of a class of vehicles in which the range covered in the cruise phase of the mission profile is most significant.

The cruise range is obtained in general by numerical integration of the vehicle trajectory equations from initiation to termination of cruise. However, if it is assumed during cruise that: (1) flight path angle is zero, $\gamma = 0$; (2) cruise speed is constant, $\frac{dV_1}{dt} = 0$; and (3) the product $\frac{L}{D} I_{sp}$ is constant, then the cruise range R can be expressed by the familiar Breguet range equation (see Appendix A)

$$R = \frac{L}{D} I_{sp} \frac{V_1}{1 - \frac{V_1^2}{V_o^2}} \ln \frac{W_i}{W_f} \quad (1)$$

subject to the restrictions of

$$L = W \left(1 - \frac{V_1^2}{V_o^2} \right) \quad (2)$$

and

$$T = -I_{sp} \frac{dW}{dt} = D \quad (3)$$

The above version of the range equation yields great circle range over a spherical nonrotating earth.

The Breguet range equation is an appropriate performance criterion in that it incorporates three fundamental measures of vehicle efficiency (aerodynamic, propulsive, and volumetric) in one equation. The first two are easily recognized in Eq 1 as the L/D and I_{sp} respectively; however, the volumetric efficiency is implicitly related to the structural factor term ($\ln W_i/W_f$) as shown in Appendix E. Volumetric efficiency for the purpose of this investigation is the ratio of configuration volume to the enclosing surface area, V/A_w .

Vehicle Model Selection

Once the performance criterion has been selected, a vehicle model is needed to relate vehicle configuration to performance. In order to proceed with the formulation of the problem it is necessary at this point to limit further the problem by specifying the cruise Mach

number regime, combustion process, fuel, and geometrical configuration class.

Mach Number Regime and Combustion Process. The lower limit of the Mach number regime investigated ($M = 8$) was coupled to the selection of combustion process. Previous studies (Ref 5) have shown that the subsonic combustion process is more efficient below flight Mach numbers of about $M = 8$, while supersonic combustion is more efficient for the higher speeds. Mach number design points of $M = 10$ and $M = 12$ were also included in this study to determine the possibility of a cruise range increase with cruise velocity, as suggested by the velocity term in the Breguet range equation. Thus, the supersonic combustion process was utilized as the model for the Mach 8, 10, and 12 design points.

Fuel. The specification of the flight Mach number regime of $M = 8$ to $M = 12$ limited the choice of fuel for the vehicle to liquid hydrogen. The main factor in the fuel selection was the high heat sink capacity of LH_2 compared to the other candidate fuels shown in Table I.

TABLE I
Candidate Fuel Properties

| Fuel | Heat of Combustion Btu/lb | Heat Sink Btu/lb | Density lb/ft ³ | M_{max} |
|-----------------------|------------------------------|---------------------|-------------------------------|-----------|
| Liquid H ₂ | 51,600 | 6000 | 4.5 | 16 |
| Methane | 21,500 | 1100 | 26.4 | 7 |
| JP-4 | 18,600 | 165 | 50.0 | 3 |

At hypersonic speeds, it is necessary to cool actively at least the combustor of the vehicle. The higher the speed--the higher the cooling requirement. A limiting speed in a sense occurs when the fuel required for cooling (which is then used for propulsion) equals that required for stoichiometric combustion. A qualitative estimate of the limiting speed can be obtained by equating the heat sink of the fuel to

the total enthalpy of the air flow (Ref 6). The last column in Table I indicates the maximum cruise Mach number for this condition. From the table it can be seen that only LH_2 has enough cooling capacity for the Mach number range considered in this investigation.

The density of LH_2 is approximately the same as the density of passenger compartments of transport type vehicles; therefore, to the first order, passengers and fuel can be interchanged. Since passenger type cruise vehicles were considered in this investigation, configurations optimized for maximum cruise range with a given payload will be the same as configurations optimized for maximum payload over a given cruise range.

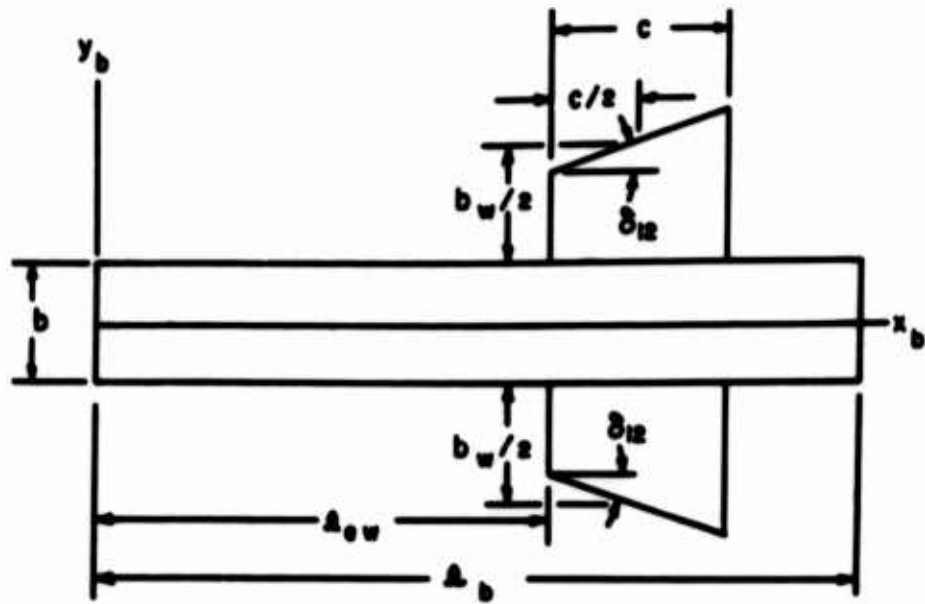
Geometrical Class. One of the major steps in the formulation of the optimum cruise vehicle problem is the selection of a geometrical class of vehicle shapes to optimize. The two-dimensional wedgelike wing body class of vehicles was selected as the model for this investigation for several reasons. Since one of the basic goals of the study was to relate vehicle configuration directly to the flow field caused by volume, lift, and heat addition, the generalized model must be amenable to these flow field calculations. Two classes of vehicle models which are amenable to the required flow field calculations are a two-dimensional wedgelike class and an axisymmetric conelike class. In an investigation of hypersonic lifting bodies which included both of the classes, Hankey found that when sharp leading edges could be maintained the two-dimensional wedgelike class had superior aerodynamic and volumetric efficiencies--two of the factors in the performance criterion of this investigation (Ref 7). The cooling capacity of the hydrogen fuel allows regenerative cooling of the leading edges of the two-dimensional configuration which permits the leading edges to be aerodynamically "sharp." Thus, on the basis of potential performance considerations the wedgelike class was selected over the conelike class as the geometrical model for this investigation.

A final consideration in the selection of the model was from the standpoint of the evolution of the optimization model level in the solution of optimization problems. Williams (Ref 8) considered a wedgelike geometrical model at a lower level of solution. Some results of the present investigation will be compared with the work of Williams to gain insight into the effect of the level of solution on the realization of results in the optimization process.

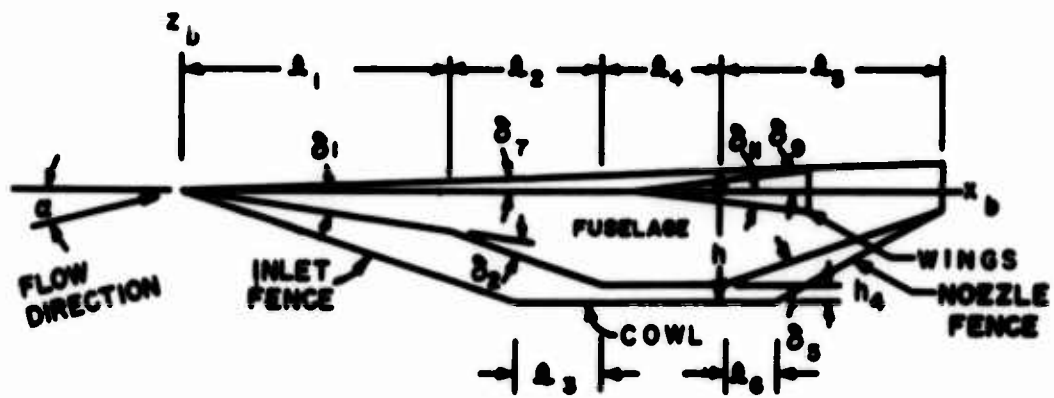
Due to the numerical nature of the algorithm for computing the flow field surrounding the vehicle, variational optimization methods, such as the calculus of variations, were not convenient to apply to this problem. Therefore, a generalized configuration model was defined by discrete parameters. Thus, the optimization problem was transformed from a variational problem to a discrete or static problem (Ref 4). The generalized configuration and the geometric parameters used to define the configuration are shown in Figure 1. The parameters chosen to define the geometrical model were balanced by two factors: (1) the desire to have many design parameters in order to keep the model as general as possible; and (2) the knowledge that each additional design parameter increases the number of possible combinations of design parameters which complicates the search for the optimum.

Force Prediction Model

At this point, methods are needed to relate the vehicle model to the performance criterion. The first two terms of the performance criterion, the L/D and I_{sp} , can be determined from a knowledge of the flow field surrounding the vehicle and the amount of hydrogen fuel added to the propulsive stream. A unique feature of the present investigation is that a complete inviscid flow field solution was performed for each change in configuration geometry. Thus, any interaction between components of the vehicle was accounted for in a fundamental manner. Although described in more detail in Appendix E, a summary of the model used to obtain the aerodynamic and propulsive forces on the



PLANFORM VIEW



PROFILE VIEW

FIG 1 Generalized Shape

vehicle is presented here. In describing the model, it is convenient to divide the vehicle into components: inlet-forebody, combustor, nozzle-aftfuselage, fuselage, and wing.

Inlet-forebody. The inlet-forebody is a two-dimensional double ramp inlet with a three-shock wave external compression system. Fences (Fig 1) extend from the nose of the vehicle to the leading edge of the inlet cowl to contain the inlet airflow. Thus, the vehicle is a wave rider (Ref 9) in the sense that the forebody "rides" on the contained plane shock waves. Forces on the inlet-forebody were obtained using the oblique shock relations with skin friction superimposed.

The geometry of the inlet was constrained to produce shock on the cowl lip, thus all of the air flow deflected by the forebody is captured by the inlet cowl and is used in the combustion process as depicted schematically in Figure 2. The inlet geometry was also constrained to produce a combustor entrance temperature of at least 2000R corresponding to the auto-ignition value of the hydrogen fuel-air mixture. The method of incorporation of these constraints in the mathematical model is discussed in the section on constraints.

Combustor. Length of the combustor and the midsection of the vehicle was defined by the length l_4 as shown in Figure 1. The combustor inviscid flow field was computed using a constant area one-dimensional supersonic combustion process. Turbulent skin friction was superimposed on the inviscid solution to determine the combustor duct forces, as it is shown in Appendix E that combustor drag does not appreciably affect the inviscid solution.

Nozzle-aftfuselage. The underside of the aftfuselage of the vehicle serves as an expansion surface for the combustor gases thereby forming, along with the nozzle cowl and fences, a two-dimensional asymmetric nozzle or half nozzle as shown in Figure 1. The nozzle forces were determined by using the method of characteristics solution to determine the pressure forces to which skin friction was superimposed.

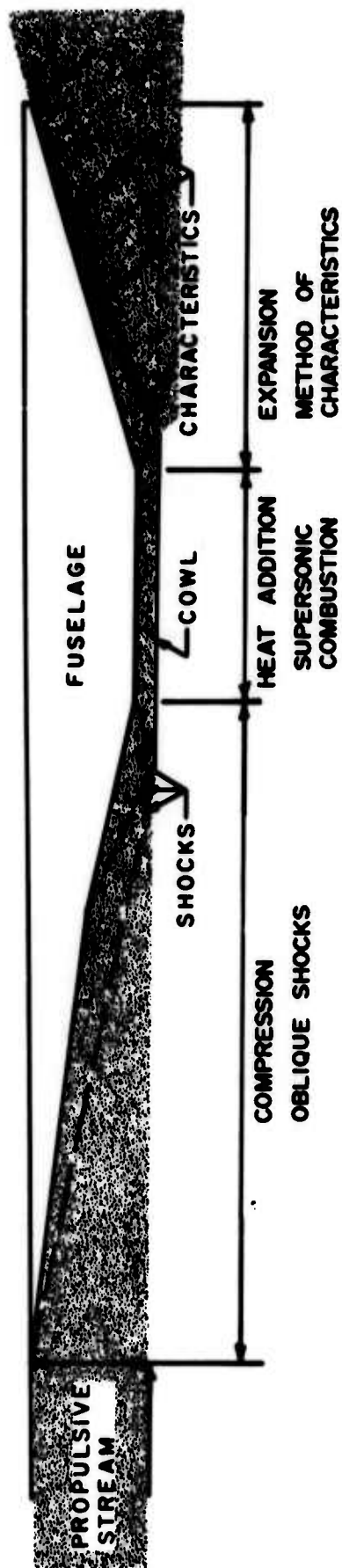


FIG 2 Propulsive Flow Field Model

Fuselage and Wing. Oblique shock or Prandtl-Meyer relations and superimposed skin friction were used to determine the forces on the top surface of the fuselage, underside of the cowl, and wing surfaces depending on the orientation of the surfaces with respect to the free-stream flow direction. Base pressure drag and leading edge drag of the fuselage and wings were included in the summation of vehicle forces. Regenerative cooling requirements of the sharp leading edges of the vehicle (one tenth of an inch diameter) were also determined for the model.

Placement of wings on the fuselage was determined by the requirement that the pitching moment about the center of gravity (assumed to be at centroid of the fuselage profile area) be zero. Rolling and yawing moments are zero due to symmetry of the model. Thus, although stability and control devices (reaction controls, thrust vectoring, and control surfaces) were not included in the analysis, the vehicle model can be characterized as being trimmed about all three axes.

In Appendix E, it is shown that in order to obtain the flow field solution it was necessary to specify the cruise Mach number M_1 , ambient temperature T_1 , and pressure p_1 at cruise altitude, as well as the amount of fuel added to the propulsive stream which can be expressed in terms of the equivalence ratio ϕ . These quantities (M_1 , T_1 , p_1 , ϕ), in addition to the vehicle geometry parameters, became variables in the problem, although not necessarily independent variables as will be explained in the section on constraints.

The problem formulation thus far has accounted for relating vehicle geometry directly to the first three terms of the performance criterion which are collectively referred to as the range factor RF given by

$$RF = \frac{L}{D} I_{sp} \frac{V_1}{1 - \frac{V_1^2}{V_o^2}} \quad (4)$$

The range factor is an indication of the range potential of a vehicle from a propulsive and aerodynamic point of view. The range factor is meaningful, however, only if the aerodynamic and propulsive characteristics are calculated in light of the constraint that the configuration provides a volume for payload, fuel, and equipment. This constraint is contained implicitly in the last term of the performance criterion--the structural factor $\ln(W_i / W_f)$. A method is needed at this point in the formulation to relate vehicle geometry to the structural factor and hence, to the performance criterion.

Structural Factor

The structural factor SF for the cruise segment of the trajectory can be written as

$$SF = \ln \frac{W_i}{W_f} = \ln \left[1 + \frac{\frac{W_{fc}}{W_{fl}} \frac{W_{fl}}{W_{to}}}{\frac{W_p}{W_{to}} + \frac{W_e}{W_{to}} + \frac{W_s}{W_{to}}} \right] \quad (5)$$

where

$$\begin{aligned} W_{fc} / W_{fl} &= \text{cruise fuel fraction} \\ W_{fl} / W_{to} &= \text{total available fuel fraction} \\ W_p / W_{to} &= \text{payload fraction} \\ W_e / W_{to} &= \text{equipment fraction} \\ W_s / W_{to} &= \text{structural weight fraction} \end{aligned}$$

The objective here is to relate the volumetric efficiency of a configuration to the structural factor. It is shown in Appendix E that this can be done by assuming the structural weight W_s to be a function of the surface area

$$W_s = \gamma_s A_w \quad (6)$$

and by assuming typical values for a set \bar{S} of structural parameters which become reference parameters for the evaluation of the structural

factor. The set consists of three weight fractions which characterize the mission of vehicles, material densities which relate weight to vehicle volume, and the proportionality constant γ_s which relates area to structural weight

$$\bar{S} = \left\{ \begin{array}{l} \frac{W_p}{W_{to}} = .05, \quad \frac{W_e}{W_{to}} = .15, \quad \frac{W_{fc}}{W_{fl}} = .5, \quad \rho_p = 4.5 \text{ lb/ft}^3, \\ \rho_{fl} = 4.5 \text{ lb/ft}^3, \quad \rho_e = 180 \text{ lb/ft}^3, \quad \gamma_s = 7.5 \text{ lb/ft}^2 \end{array} \right\} \quad (7)$$

The assumed values of the structural set members are characteristic of hydrogen fueled passenger transport type vehicles (Ref 10).

Appendix C contains a range performance sensitivity analysis of each member of the structural set.

The main purpose of forming the structural set \bar{S} , however, was not to evaluate the absolute level of performance but rather to evaluate the change in the performance as a function of the volumetric efficiency of the configurations. This can now be done. Thus, three basic measures of vehicle efficiency (aerodynamic, propulsive, and volumetric) are incorporated in the performance criterion.

Constraints

Constraints incorporated in the mathematical formulation can either help or hinder the optimization process. Constraints are helpful if the information from the constraints can be used to calculate configuration variables thereby reducing the number of search variables. Besides reducing the number of search variables, it will be shown in the next chapter that constraints can provide explicit information on the boundaries of the feasible region of the design space.

Constraints complicate the optimization process if they cannot be used to reduce the design space and must, for example, be adjoined to the performance criterion to form a penalized performance

criterion (Ref 11). Considerable effort was made in this investigation, therefore, to use constraint information to limit the dimensions and feasible region of the design space. The two general types of constraints incorporated in the mathematical formulation were the equality and inequality constraints.

Equality. Relationships derived from equality constraints can be used to reduce the number of search variables as shown in Appendix E. In addition to geometric interrelationships, the following equality constraints were imposed on the formulation: equilibrium flight, from selection of performance criterion; trimmed condition, to provide a gross measure of longitudinal stability; ambient pressure-temperature relationship, from the physics of the atmosphere; and shock on inlet cowl lip, from ramjet inlet design practice to give optimum performance. The shock on inlet lip constraint also reduces the dimensions of the design space which still contains the optimum. Two nongeometric parameters, combustor entrance temperature T_4 and takeoff weight W_{t0} , were also introduced to replace two geometric parameters, δ_2 and l_1 , (see Fig 1) as design variables. The motivation for the substitution becomes apparent later when discussing inequality constraints. Although the substitution of design variables added two parameters to the number necessary to define the generalized configuration, two equality constraint relations were also added keeping the effective number of independent variables constant.

Inequality. Equality constraints have a direct influence on all design points in the design space whereas inequality constraints influence the design only when the constraint is encountered. Two forms of inequality constraints were present in the problem investigated in this study. One form was applied directly as an upper and or lower bound on the design variable. A lower bound of 2000R (auto-ignition value of fuel-air mixture) was imposed on the combustor entrance temperature. An upper bound of 500,000 lb was placed on the takeoff weight as a

typical value since the results will show range performance to increase monotonically with takeoff weight. Inequality constraints placed directly on the design variables are easily handled. In fact, they are even desirable from an optimization search standpoint since they explicitly define the boundaries of the feasible region of the design space.

On the other hand, inequality constraints which are functions of the independent design variables are more difficult to handle. The inequality constraint functions are calculated along with the performance criterion. If the value of the constraint function exceeds the specified amount, then the value of the performance criterion for that set of design variables is penalized. Thus, the function type inequality constraint also limits the feasible region of the design space, although after the fact since the numerical calculations have already been performed.

The following functional type inequality constraints were incorporated in the mathematical formulation:

- (1) A cooling constraint, such that the fuel required for regenerative cooling (see Appendix D) was less than or equal to fuel required for propulsion.
- (2) A wing placement constraint (see Fig 1) such that the attachment point of the leading edge of the wing was on the fuselage ($l_{ew} < l_b$) and in the xy-plane of the body axis system ($\delta_7 \geq 0$) for trimmed flight.
- (3) A wing area constraint, such that planform area was greater than or equal to zero ($S_p \geq 0$). Since in this formulation (see Appendix E) the wing area was directly proportional to thrust minus fuselage drag, a wing area greater than zero indicates constant velocity flight can be maintained. Notice that the wing area of the optimized configurations will reflect the cruise point. Takeoff and landing considerations, not included in this investigation, would perhaps require the lower bound on the wing area constraint to be increased.

The problem formulation contained in this chapter reflects one level of solution of the general problem of configuration optimization of hypersonic vehicles. The selections and assumptions made during the formulation of the problem have resulted in a particular class of configurations to be optimized for a specific set of conditions. Since the mathematical formulation of the problem does not include all classes of configurations and a consideration of all constraints, limitations, and influences; the optimized configuration is not expected to offer a final solution to the optimum vehicle problem. The present formulation of the problem is expected, however, to contribute to a final solution by answering some fundamental questions concerning the performance and configuration trends of optimum cruise vehicles when vehicle geometry is related directly to the surrounding flow field.

Appendix E contains the development and a listing of the Fortran IV computer program which resulted from the problem formulation discussion presented in this chapter. The program relates the performance criterion (cruise range R) to the parameterized configuration shown in Figure 1. So that functionally

$$R = f(\bar{X}) \quad (8)$$

where \bar{X} is a set of independent design variables consisting of

$$\bar{X} = (\delta_1, T_4, \phi, l_4, \delta_5, l_5, l_6, \delta_9, \delta_{11}, \delta_{12}, AR, b, \delta_7, \alpha, W_{t0}, M_1)$$

Due to the numerical methods necessary to predict the flow field and to integrate the forces on the vehicle, the algebraic form of Eq 8 is not known. However, the value of the performance criterion can be determined numerically for values of the independent variables. Thus the relationship between the performance criterion and independent design variables is characterized as computational as opposed to mathematical. This distinction becomes important in the selection of optimizing methods which is discussed in the next chapter.

III. Optimizing Method

Introduction

Once the problem is formulated, the next step in the optimization process is the selection of an optimizing method. The methods can be classified into two groups--indirect and direct. Indirect methods are mathematical in the sense that necessary conditions are used to find the extremum. In ordinary maxima and minima theory, the value of x is sought which causes the first derivative of $f(x)$ with respect to x to vanish, or in the case of the calculus of variations the curve is sought which satisfies the Euler necessary condition. In the direct method, on the other hand, the value of x which makes $f(x)$ an extremum is determined by direct comparison of the value of the function at two or more points in the operating space (Ref 12). Problems formulated by the indirect methods can be solved either numerically or analytically; whereas, the direct method implies a numerical solution. With the advent of the digital computer, the direct method of optimization has become practical.

The cruise vehicle optimization problem has been formulated so that the performance criterion R can be evaluated for values of the independent design variables \bar{X} by the computational algorithm contained in Appendix E. The objective now becomes to determine the combination of independent design variables \bar{X}_{opt} which maximizes the cruise range R

$$R_{max} = f(\bar{X})_{opt} = f(X_1, X_2, \dots, X_{15})_{opt} \quad (9)$$

for design cruise Mach numbers of 8, 10, and 12.

In order to understand the factors involved in the direct search for the maximizing set of design variables, it is helpful to introduce some geometric concepts of the design variable space and response surface such as those discussed by Wilde (Ref 12). The design variable plane and response surface are schematically depicted in Fig 3

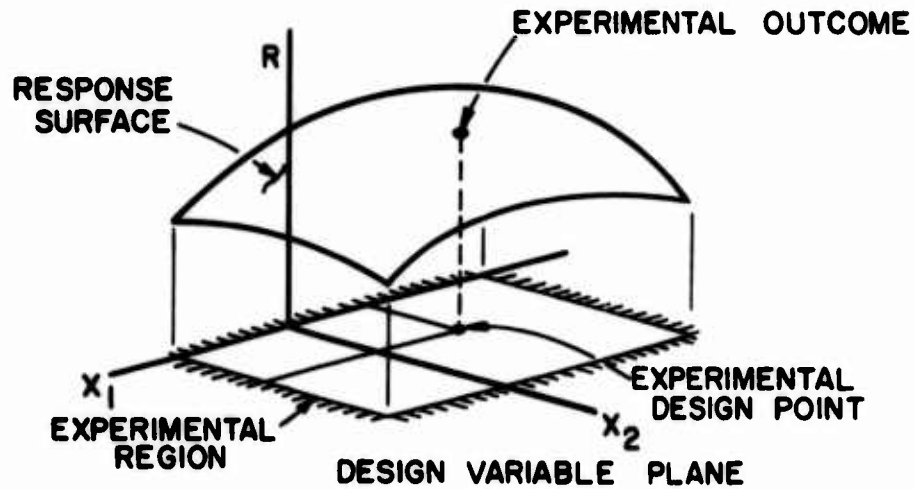


FIG 3 Graphical Representation of Direct Search Problem for Two Design Variables

for a function of two independent variables. In the case of configuration optimization, for example, each point in the design variable plane represents a configuration and comprises the input for a possible numerical experiment. The point above it on the response surface (the value of the function) represents the experimental outcome. The feasible values which each of the independent variables can assume form a bounded region in the design variable plane. This bounded region is referred to as the experimental region or design space.

Geometrically, the direct search method of function optimization locates the summit of the response surface by performing numerical experiments with points contained in the design space. The methods used to select the points with which to perform the numerical experiments are called search techniques which are discussed later. Although impossible to represent physically for functions of more than two independent variables, the concepts of the design space and response surface can be extended to functions of n independent variables.

Optimum Configuration Search

The search for the optimum cruise vehicle configuration can be divided into three phases. The first phase is the definition of the design space or experimental region from the design variable plane; the second phase is the systematic exploration of the design space for the optimum; and the third phase is the termination of the search.

Design Space Definition. The experimental region or design space is the portion of the design variable plane which contains the candidate optimum configurations. The upper and lower bounds on each of the design variables (X_1, \dots, X_{15}) define the design space. Inequality constraints on the independent variables and care to include the true optimum configuration in the design space were factors in placing upper and lower bounds on the 15 design variables comprising the design space. The minimum combustor inlet temperature T_4 is an example of a design space boundary formed by an inequality constraint. Initially upper and lower bounds on the unconstrained design variables are chosen such that the design space represents all feasible configurations. As knowledge of the response surface is gained, the size of the design search space which still contains the optimum configuration can be reduced. Once the design space is defined, multivariable search techniques are used to systematically explore the design space for the optimum configuration at the specified design Mach number.

Direct Search Algorithms. Several well known direct search algorithms (Table II) have been combined into a single computer program called the Automated Engineering and Scientific Optimization Programs - AESOP (Ref 14). The program AESOP is independent of the problem being solved. In aerodynamic configuration problems, for example, the computer program containing the algorithm for the evaluation of the configuration performance is linked to AESOP to perform a systematic search for the optimum configuration. In the cruise vehicle problem, for a given design Mach number M_1 , points

TABLE II

Basic Search Algorithms Contained in Program AESOP

1. Sectioning - Succession of one-dimensional optimization calculations parallel to coordinate axes. Variables may be perturbed in random or natural order.
2. Pattern - A Ray Search in the gross direction defined by a previous search or search combination.
3. Magnification - Straightforward magnification or diminution about the origin.
4. Steepest Ascent - Search along the weighted gradient direction. Several weighting options available.
5. Adaptive Creeping - Search in small incremental steps parallel to the coordinate axes. Step-size adjusted automatically in the algorithm. Variables may be perturbed in random or natural order.
6. Quadratic - Second-order multivariable curve fit to the function being optimized, followed by search in direction of second-order surface optimum.
7. Davidon's Method - An attempt to achieve the advantages of second-order search from an ordered succession of first-order (steepest ascent) searches.
8. Random Point - Function to be optimized is evaluated at a set of uniformly distributed random points in a specified region.
9. Random Ray Search - Function is optimized by search along a sequence of random rays having a uniformly distributed angular orientation in the multivariable parameter space.

from the design space were selected by a specified search algorithm in AESOP and evaluated by the vehicle Design and Evaluation Computer Program - DECP (Appendix E) until the maximum cruise range performance and corresponding configuration were obtained.

Although nine search algorithms were available in AESOP, it became evident in early searches that search algorithms (steepest ascent, quadratic, and Davidon's method) requiring derivatives of the performance function with respect to each of the search variables were too time consuming. For example, since the derivatives must be determined numerically, to find the gradient of the performance function at one point on the response surface of a 15 search variable problem required 30 evaluations by DECP. This required approximately 10 minutes of 7094 computer time. Furthermore, in the steepest ascent method, the path of steepest ascent and the positive gradient of the performance function coincide only if unit perturbations of each of the search variables produce a similar change in the performance function (Ref 14). In the case of the cruise vehicle problem, the effect on the performance function of a unit perturbation of the equivalence ratio $\Delta\phi = 1$ is large; whereas, a unit perturbation of the takeoff weight $\Delta W_{t0} = 1$ is negligible. A weighting matrix can be chosen in an effort to account for varying effects of a unit perturbation on the performance function. The necessity of choosing a weighting matrix in order to determine the true path of steepest ascent, however, makes it questionable if the computational time required to determine accurately the gradient of the performance function was used most effectively. Search algorithms available in AESOP which produced satisfactory results for the cruise vehicle problem were the random point, sectioning, and adaptive creep algorithms.

In the random point method, a series of design points are selected from points which have uniform distribution throughout the design space. The performance at these design points is evaluated one by one and the design point with the highest value of the performance

function is retained. This method has an advantage in that its effectiveness is independent of the shape of the response surface. The method works as well on multimodal as on unimodal (single peak) response surfaces. One disadvantage of the method is that, due to the size of the design space, for most problems many points must be evaluated before attaining a high probability that the best point selected is actually at or near the optimum. The random point method was used in this investigation during the early exploration of the design space when the response surface of the criterion function was unknown.

Another method which is useful in the early phases of the optimum configuration search is sectioning. Search by sectioning is a series of one-dimensional searches along the entire ray in design space parallel to each of the coordinate axes as shown in Figure 4.

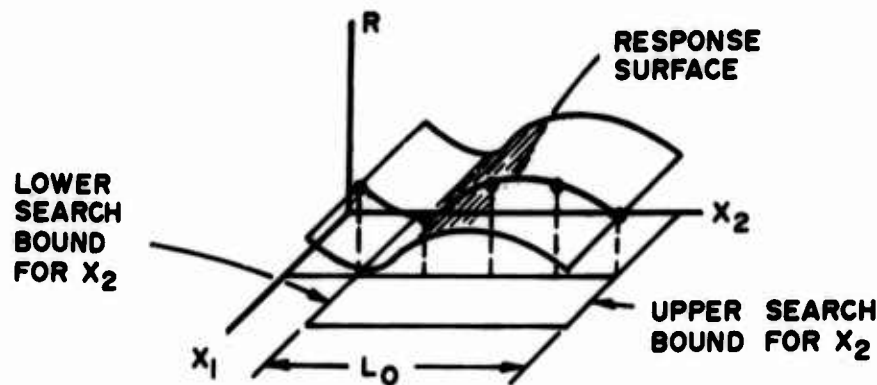


FIG 4 Sectioning Search for Extremal Along Ray Parallel to x_2 Axis

The one-dimensional ray in design space is formed by fixing all the search variables except the one on which the search is to be performed. The length L_0 of the ray is determined by the upper and lower bound of the design variable. Points are evaluated along the ray in order to find the value of the sectioned variable which results in the highest value of the performance function. The value of the variable giving

the maximum performance is retained and the process repeated for each of the remaining design variables.

The adaptive creeper search method is another form of sectioning. However, instead of searching along the entire length of the ray parallel to the coordinate axis as in the section search, only small perturbations are made in one of the independent variables. Perturbations in the independent variable are continued until no further improvement in performance is possible. When the process has been repeated for each independent variable in turn, a creeper search cycle is completed. In the case where there is no interaction between the independent variables only one search cycle is required to locate a peak in the response surface. This case is illustrated in Fig 5 for the contours of a performance function which has two independent variables. Usually, however, interaction between the independent variables is present so that more than one search cycle is required to locate the optimum. This case is also illustrated in Fig 5 where three

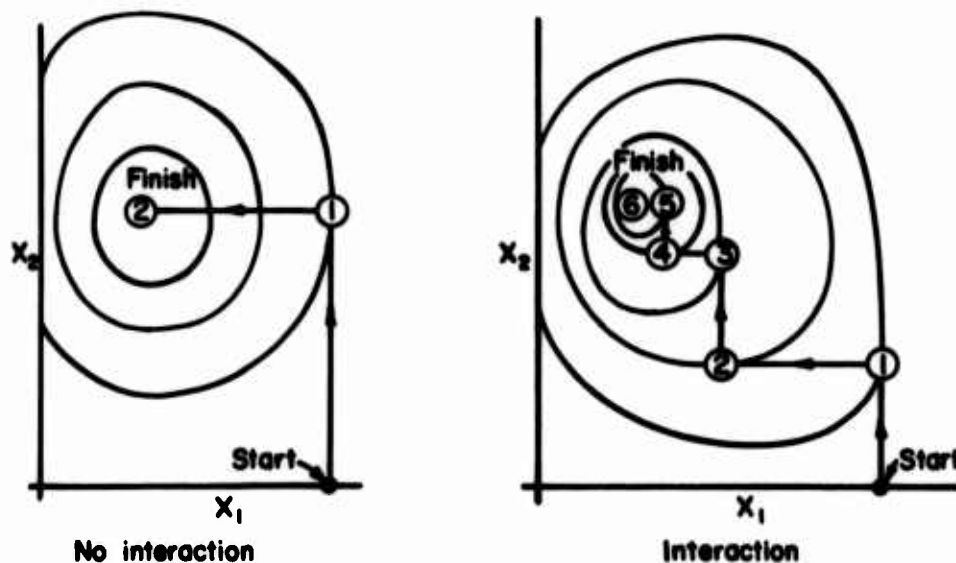


FIG 5 Creeper Search With and Without Interaction of Design Variables

adaptive creeper search cycles would be required to locate the optimum. The creeper method was effective in the cruise vehicle optimization in the latter phases of the systematic search.

Figures 6 - 9 illustrate the performance of the random point, sectioning, adaptive creeper, and steepest ascent optimum seeking algorithms on a test function J

$$J = 10 - ((X_2 - X_1)^2 + (1 - X_1)^2) \quad (10)$$

where the shape of the response surface is known. Contours of the response surface of the test function are shown in the figures as well as the starting point and search limits for all search techniques. Although the response surface is unimodal, the search problem is not a particularly easy one since a mild ridge is present in the response surface.

All searches in the above example were terminated after 50 evaluations of the performance function. Progress during the search is indicated in the figures by arrows and the number of evaluations of the performance function required to arrive at the point. The results of the example show the adaptive creeper to be most efficient while the steepest ascent is least efficient in this example. It should be pointed out, however, that the best search technique depends on design space and response surface characteristics which, of course, are a function of the particular search problem.

Termination of Search. The last step in the direct search for the optimum cruise vehicle configuration is the termination of the search. A total of approximately 3000 numerical experiments were made on a IBM 7094 computer to obtain the optimum configuration for the design cruise Mach number 8, 10, and 12 shown in the next chapter. Searches were terminated when the gain in the performance function R was less than 1% between search cycles. Searches were also started from different points in the design space to examine the possibility of a multimodal performance response surface. The response surface

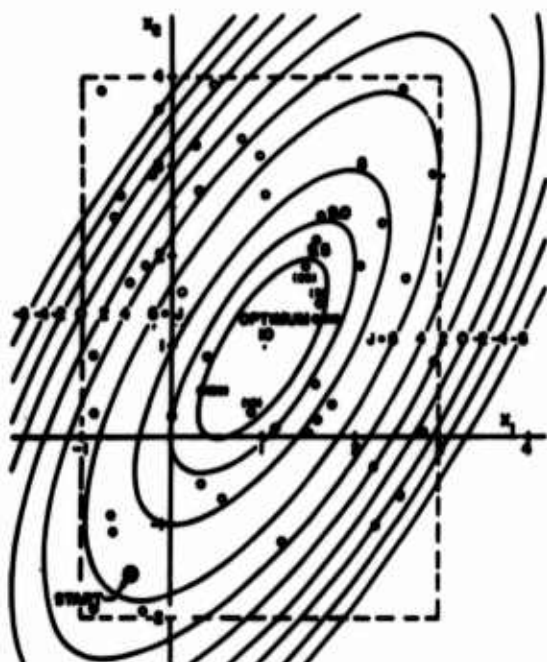


FIG 6 Random Point Search

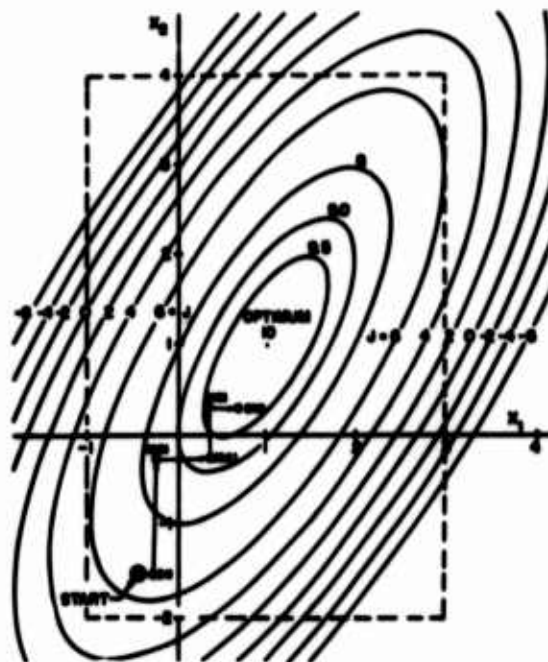


FIG 7 Sectioning Search

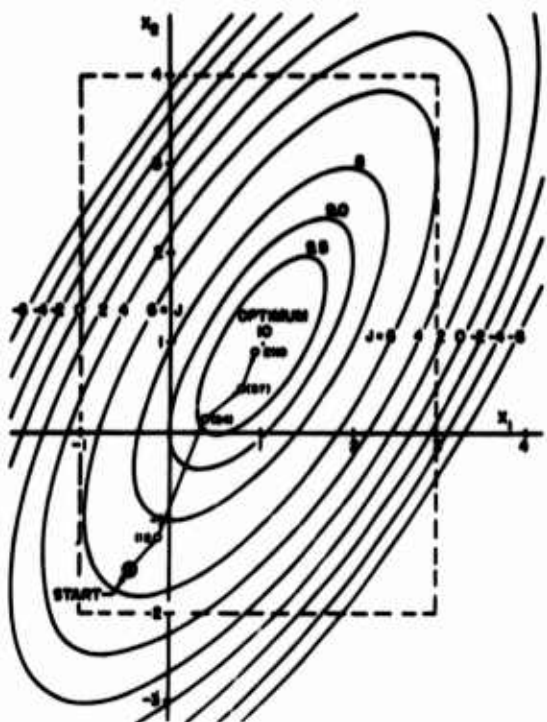


FIG 8 Adaptive Creeper Search

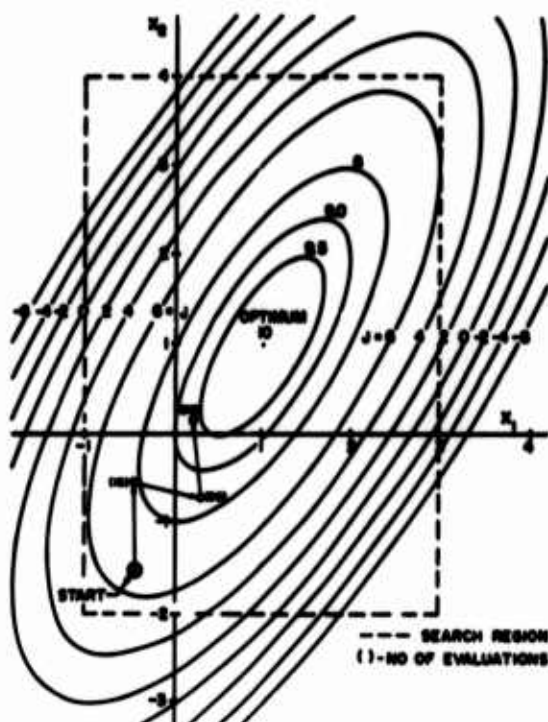


FIG 9 Steepest Ascent Search

appeared to be unimodal, however, for a large region around the optimum design point. In the region of optimum point the response surface was relatively "flat" so that a large number of configurations correspond to nearly equal performance. This is illustrated in the performance sensitivity analysis contained in the next chapter.

IV. Analysis of Results

Results of this study are presented in two parts. The first section deals with configuration characteristics of optimum range hypersonic cruise vehicles; whereas, the second section deals with performance.

Optimum Configuration Characteristics

The overall goal of this investigation was a fundamental analysis of configurations in which the optimization process included the effects of the interaction of the aerodynamic and propulsive flow fields. Table III contains the values of the independent variables which were determined by the direct search methods discussed in the previous chapter. The table also contains the dependent variables which were calculated from equality constraint relationships as developed in Appendix E. Together, the independent and dependent variables define the vehicle configuration. The fuselage of the shapes shown in Figs 10, 11, and 12 are waveriders in that the edge of the inlet fences coincides with the shock waves produced by the inlet-forebody. Thus, the forebody of the vehicle "rides" on the contained plane shock waves. Nozzle fences, also evident on the configurations maintain two-dimensional pressure distribution on the aftfuselage.

The shapes were determined by directly relating geometry to the flow field quantities surrounding the vehicles. Some of the flow field quantities associated with the propulsive stream are shown in Figs 10 - 12. The Mach number can be seen to decrease between the entrance and exit of the combustor which is characteristic of the constant area supersonic combustion process. Figures 10 - 12 illustrate that the wall static pressures at termination of the aftfuselage were above free-stream pressure for the optimum configuration. Aftfuselage length and hence nozzle expansion was terminated by the zero fuselage base constraint in the case of the Mach 10 and 12 designs. Figure 11 shows that further nozzle expansion was possible in the case of the Mach 8

TABLE III
Configuration Definition Quantities^a for Design
Mach Numbers of 8, 10, and 12

| QUANTITIES | DESIGN MACH NUMBER | | |
|--------------------------|--------------------|----------|----------|
| | 8 | 10 | 12 |
| Independent | | | |
| Inlet | | | |
| δ_1 deg | 11.36 | 8.21 | 6.40 |
| T_4 R | 2000. | 2000. | 2000. |
| Combustor | | | |
| ϕ | .449 | .530 | .795 |
| l_4/h_4 | 5.00 | 5.00 | 5.00 |
| Nozzle | | | |
| δ_3 deg | 15.15 | 14.02 | 12.92 |
| l_4/h_4 | 1.51 | 2.36 | 2.85 |
| l_3/l_1 | 1.23 | 1.29 | 1.11 |
| Wing | | | |
| δ_{11} deg | 8.80 | 7.41 | 7.18 |
| δ_9 deg | -5.32 | -5.21 | -5.25 |
| δ_{12} deg | 0. | 0. | 0. |
| AR | .345 | .240 | .245 |
| Fuselage | | | |
| α deg | .747 | 1.01 | 1.05 |
| δ_7 deg | 0. | 0. | 0. |
| W_{t0} lb | 500,000. | 500,000. | 500,000. |
| b/h | 1.21 | 1.33 | 1.36 |
| Dependent | | | |
| δ_2 deg | 5.48 | 4.81 | 4.12 |
| l_2/l_1 | .781 | .771 | .777 |
| l_3/l_1 | .219 | .206 | .204 |
| h_4/l_1 | .0390 | .0271 | .0214 |
| Sp/bl_1 | .374 | .306 | .239 |
| l_{ew}/l_1 | 3.20 | 3.17 | 2.99 |
| bw/l_1 | .273 | .185 | .148 |
| c/l_1 | .791 | .769 | .606 |
| p_1 lb/ft ² | 13.73 | 12.10 | 9.44 |
| l_1 ft | 52.84 | 64.41 | 76.12 |
| T_1 R | 438.3 | 443.2 | 453.0 |

^a Data obtained with structural parameter set S values of: $W_p/W_{t0} = .05$
 $W_e/W_{t0} = .15$, $W_{fc}/W_{fl} = .5$, $\rho_p = 4.5 \text{ lb/ft}^3$, $\rho_f = 4.5 \text{ lb/ft}^3$,
 $\rho_e = 180 \text{ lb/ft}^3$, $\gamma_s = 7.5 \text{ lb/ft}^2$.



■ - INLET, NOZZLE FENCE

| <u>Performance (cruise)</u> | | | <u>Propulsive</u> | | |
|-----------------------------|---------|--------------------|---------------------------------------|--------|-----------------|
| Altitude (Initial) | 112000. | ft | Inlet | | |
| Range | 4991. | nm | capture area | 799.99 | ft ² |
| Velocity | 4861. | nm/hr | entrance | | |
| Breguet Range factor | 9878. | nm | pressure | .00649 | atm |
| <u>Structural</u> | | | Mach | 8. | |
| Fuselage Dimensions | | | exit | | |
| length | 169.79 | ft | pressure | .444 | atm |
| height | 25.17 | ft | Mach | 3.18 | |
| width | 30.46 | ft | Combustor | | |
| Weight | | | entrance | | |
| takeoff | 500000. | lb | pressure | .444 | atm |
| fractions | | | Mach | 3.18 | |
| payload W_p / W_{to} | .050 | | temperature | 2000. | R |
| structural W_s / W_{to} | .258 | | exit | | |
| fuel W_{f1} / W_{to} | .542 | | pressure | 1.273 | atm |
| equipment W_e / W_{to} | .150 | | Mach | 1.84 | |
| Breguet Structural | | | temperature | 4475. | R |
| factor $\ln (W_i / W_f)$ | .465 | | specific heat ratio | 1.269 | |
| <u>Aerodynamic</u> | | | Nozzle-aftfuselage | | |
| Angle of attack | .747 | deg | exit pressure ratio $\frac{P_2}{P_1}$ | 1.83 | |
| Dynamic pressure | 602. | lb/ft ² | Equivalence ratios | | |
| Lift-drag ratio | 2.453 | | propulsive | .449 | |
| (power on) | | | coolant | .125 | |
| | | | Specific impulse | 2678. | sec |

FIG 10 Configuration and Characteristics of a Hydrogen-Fueled Mach 8 Cruise Vehicle



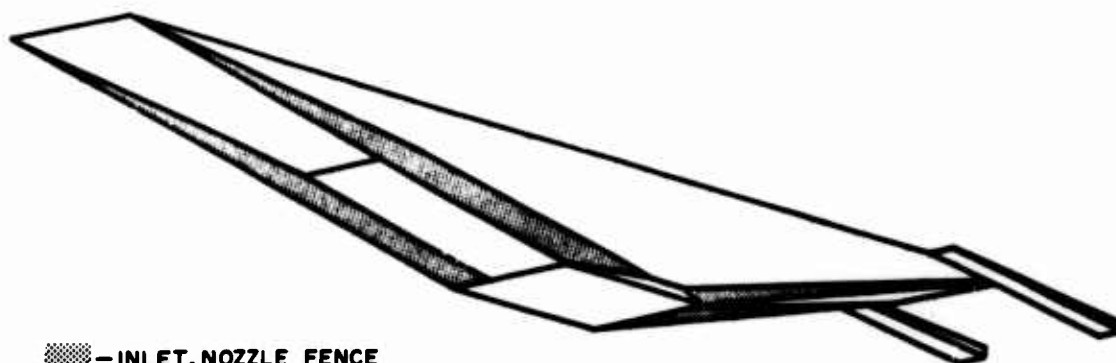
INLET, NOZZLE FENCE

| <u>Performance (Cruise)</u> | | | <u>Propulsive</u> | | |
|-----------------------------|---------|-------|---------------------------------------|---------|-----------------|
| Altitude (initial) | 118000. | ft | Inlet | | |
| Range | 4777. | nm | capture area | 727. 71 | ft ² |
| Velocity | 6110. | nm/hr | entrance | | |
| Breguet Range factor | 11191. | nm | pressure | . 00572 | atm |
| | | | Mach | 10. | |
| | | | exit | | |
| | | | pressure | . 3944 | atm |
| | | | Mach | 4. 28 | |
| | | | Combustor | | |
| | | | entrance | . 3944 | atm |
| | | | pressure | 4. 28 | |
| | | | Mach | | |
| | | | temperature | 2000 | R |
| | | | exit | | |
| | | | pressure | 1. 12 | atm |
| | | | Mach | 2. 56 | |
| | | | temperature | 4630. | R |
| | | | specific heat ratio | 1. 266 | |
| | | | Nozzle-aftfuselage | | |
| | | | exit pressure ratio $\frac{P_e}{P_i}$ | 1. 77 | |
| | | | Equivalence ratios | | |
| | | | propulsive | . 530 | |
| | | | coolant | . 218 | |
| | | | Specific impulse | 2053. | sec |

| <u>Structural</u> | | |
|---------------------------|------------|----|
| Fuselage Dimensions | | |
| length | 203. 90 ft | |
| height | 22. 52 ft | |
| width | 29. 96 ft | |
| Weight | | |
| takeoff | 500000. | lb |
| fractions | | |
| payload W_p / W_{to} | . 090 | |
| structural W_s / W_{to} | . 204 | |
| fuel W_{fz} / W_{to} | . 516 | |
| equipment W_e / W_{to} | . 190 | |
| Breguet Structural | | |
| factor $f_n (W_i / W_f)$ | . 427 | |

| <u>Aerodynamic</u> | | |
|--------------------|-----------|--------------------|
| Angle of attack | 1. 01 deg | |
| Dynamic pressure | 847. | lb/ft ² |
| Lift-drag ratio | 2. 70 | |
| (power on) | | |

FIG 11 Configuration and Characteristics of a Hydrogen-Fueled Mach 10 Cruise Vehicle



| <u>Performance (cruise)</u> | | | <u>Propulsive</u> | | |
|-----------------------------|---------|-------|---------------------------------------|--------|-----------------|
| Altitude (initial) | 121000. | ft | Inlet | | |
| Range | 4388. | nm | Capture area | 668.98 | ft ² |
| Velocity | 7413. | nm/hr | entrance | | |
| Breguet Range factor | 10688. | nm | pressure | .00460 | atm |
| | | | Mach | 12. | |
| | | | exit | | |
| | | | pressure | .2985 | atm |
| | | | Mach | 5.36 | |
| | | | Combustor | | |
| | | | entrance | | |
| | | | pressure | .2985 | atm |
| | | | Mach | 5.36 | |
| | | | temperature | 2000. | R |
| | | | exit | | |
| | | | pressure | .9881 | atm |
| | | | Mach | 3.01 | |
| | | | temperature | 5015. | R |
| | | | specific heat ratio | 1.262 | |
| | | | Nozzle-aftfuselage | | |
| | | | exit pressure ratio $\frac{P_a}{P_i}$ | 2.34 | |
| | | | Equivalence ratios | | |
| | | | propulsive | .795 | |
| | | | coolant | .339 | |
| | | | Specific impulse | 1314. | sec |

| <u>Structural</u> | | |
|---------------------------|---------|----|
| Fuselage Dimensions | | |
| length | 228.56 | ft |
| height | 21.14 | ft |
| width | 28.76 | ft |
| Weight | | |
| takeoff | 500000. | lb |
| fractions | | |
| payload W_p / W_{to} | .050 | |
| structural W_s / W_{to} | .296 | |
| fuel W_{ff} / W_{to} | .504 | |
| equipment W_e / W_{to} | .150 | |
| Breguet Structural | | |
| factor $\ln(W_i / W_f)$ | .410 | |

| <u>Aerodynamic</u> | | |
|--------------------|------|--------------------|
| Angle of attack | 1.05 | deg |
| Dynamic pressure | 951. | lb/ft ² |
| Lift-drag ratio | 3.05 | |
| (power on) | | |

FIG 12 Configuration and Characteristics of a Hydrogen-Fueled Mach 12 Cruise Vehicle

design since the fuselage base is not zero; however, the additional nozzle forces which would be obtained by further expansion did not compensate for the additional structural weight associated with the increased fuselage length.

Takeoff weight, which was an independent design variable in this study, and members of the structural set \bar{S} listed in Table III determined the scale or dimension of the vehicles shown in Figures 10 - 12. The cruise range also depends explicitly on the assumed values of the set members; therefore, a range sensitivity analysis with respect to each member of the set is contained in Appendix C. The sensitivity analysis indicated that the cruise range either increased or decreased monotonically in the investigated domain of the set members. Payload density ρ_p , for example, did not have a natural optimum--higher payload density produced higher cruise range. While the cruise range depends explicitly on the values assumed for the structural set \bar{S} , the range factor shown in Fig 10 - 12 does not. The range factor will be considered more fully when discussing the performance aspects of the vehicle. First, however, the configuration characteristics of the vehicle will be examined.

Insight into why a particular configuration was optimum can be gained by exploring the design space in the vicinity of the optimum point. Each figure in the series of Figs 13 - 14 and Figs 16 - 28 illustrates, for the Mach 10 configuration, the influence of the independent design variables on the range factor, structural factor, and cruise range R . The variation of cruise range R with each of the independent variable represents one-dimensional "cuts" in the response surface around the optimum point. Aerodynamic and propulsive efficiencies variations are reflected by the range factor; whereas the volumetric efficiency is reflected by the structural factor of the Breguet range equation repeated here for convenience

$$R = \frac{L}{D} \text{Isp} \overbrace{\frac{V_1}{1 - V_1^2 / V_0^2}}^{\text{range factor}} \overbrace{\ln (W_i / W_f)}^{\text{structural factor}} \quad (1)$$

Figures 13, 14 and 16 through 28 were generated by perturbing one independent design variable at a time while holding the other independent quantities (except nozzle length and equivalence ratio) at the optimum values for the Mach 10 design. The nozzle length was automatically reoptimized during the solution of Eq 1 and the equivalence ratio was adjusted when necessary to provide additional thrust to satisfy the equilibrium flight constraint (thrust equal drag). In discussing some of the characteristics of the optimum configuration, it is convenient to divide the vehicle into components: inlet-forebody, combustor, nozzle-aftfuselage, wing, and fuselage.

Inlet-forebody. Two independent design variables define the shape of the inlet-forebody. One is the inlet ramp angle δ_1 which, in addition to defining the profile of the underside of the forebody, also controls the relative shock strength produced by the first and second inlet ramps. From a propulsive efficiency standpoint, the goal of inlet design is to achieve maximum pressure recovery. According to Orlov (Ref 15) maximum pressure recovery occurs for the three shock inlet of type considered in this study when the pressure rise across the first two shock waves is equal. This result was confirmed in this investigation; however, Fig 13 illustrates that, although the response surface is rather flat near the optimum point, the optimum δ_1 for maximum range was greater than the δ_1 producing maximum pressure recovery. The reason for the difference is apparent also from Fig 13 as the volumetric efficiency of the vehicle (reflected by the structural factor) increased as δ_1 increased. Thus, consideration of the volume producing aspects of the inlet-forebody resulted in a slight change from the conventional one-dimensional goal of inlet design--that of maximum inlet pressure recovery. This information could be used in future

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4269$

$R_0 = 4777$ NAUTICAL MILES

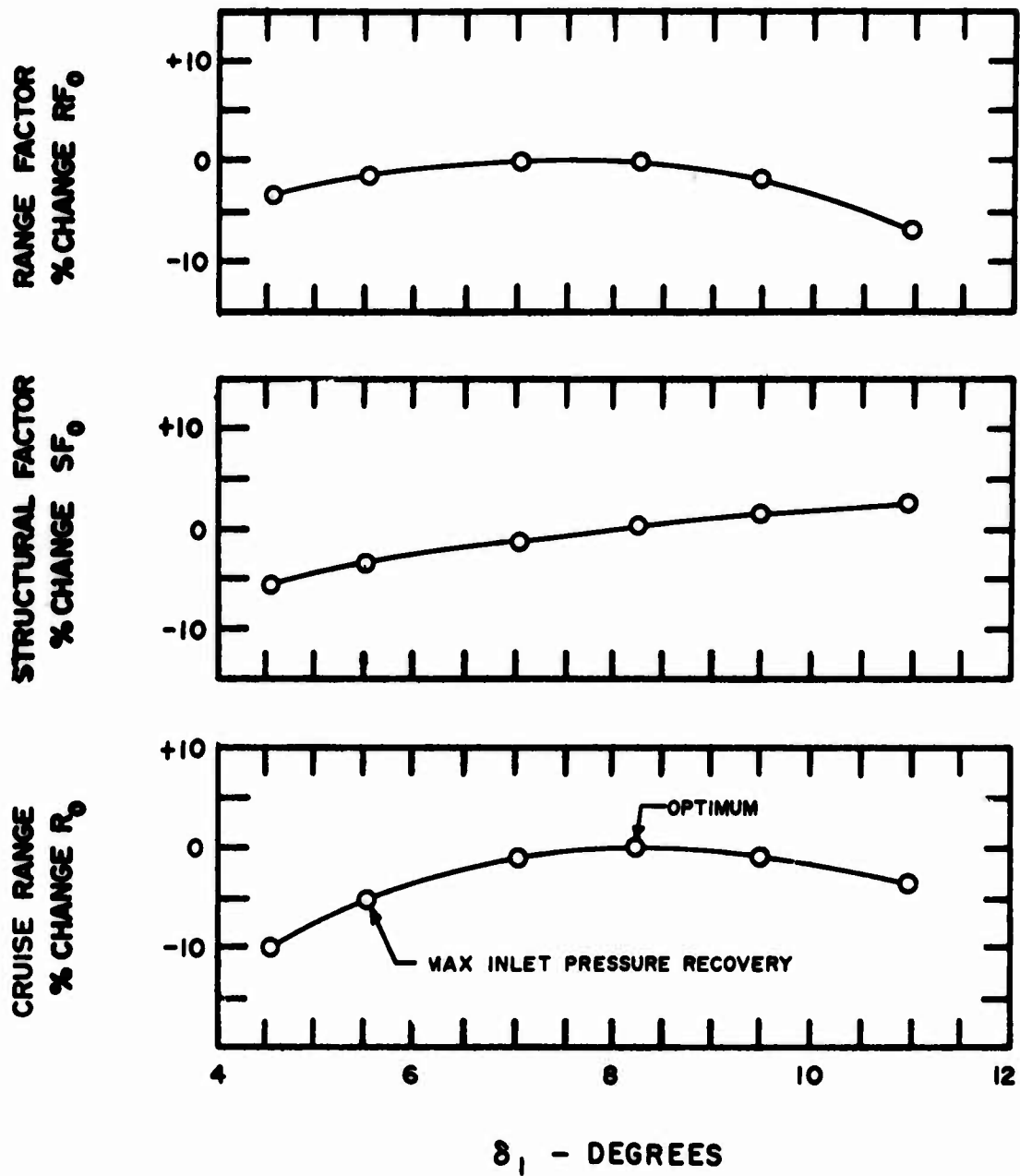


FIG 13 First Inlet Ramp Angle δ_1 Influence on Cruise Range and Factors of Breguet Range Equation

configuration optimization studies of this nature to limit the feasible region of the design space. For example, the maximum pressure recovery configuration inlet-forebody could be used as a starting point with searches in the design space limited to those inlet-forebody configurations with improving volumetric efficiency.

Whereas the first inlet-forebody design variable δ_1 defines the inlet profile and relative shock wave strengths from the inlet ramps, the second inlet design variable, combustor entrance temperature T_4 , defines the total strength of the shock wave system. A higher combustor entrance temperature requires a stronger inlet shock wave system which in turn demands a greater deflection of the propulsive stream. Thus, in terms of configuration geometry, the combustor entrance temperature defines the inlet-forebody length to thickness ratio. Figure 14 illustrates that the trade off between the structural factor and range factor with varying combustor entrance temperature resulted in a rather flat response surface; however, in the region of allowable solutions ($T_4 \geq 2000$ R) a combustor entrance temperature of $T_4 = 2000$ R produced maximum range. This is an interesting result since vehicle thickness, necessary to provide volume in the vehicle, is automatically provided by the propulsive constraint ($T_4 \geq 2000$ R autoignition temperature for hydrogen air mixture). The thickness ratio of the Mach 8, 10, and 12 designs can be observed from Fig 10 - 12 to become progressively finer as the flow deflection needed to produce the required shock strength is reduced as Mach number increases.

An interesting characteristic of the forward section of the vehicle is that it produced very little lift as shown in Figure 15. In fact, neglecting friction, at zero angle of attack and zero angle of incidence of the upper surface of the fuselage, the lift produced by the forward section of the vehicle is zero. The force in the lift direction produced by pressure acting on the underside of the forebody is cancelled by the downward pressure force acting on the inlet cowl. The zero lift result

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4269$

$R_0 = 4777$ NAUTICAL MILES

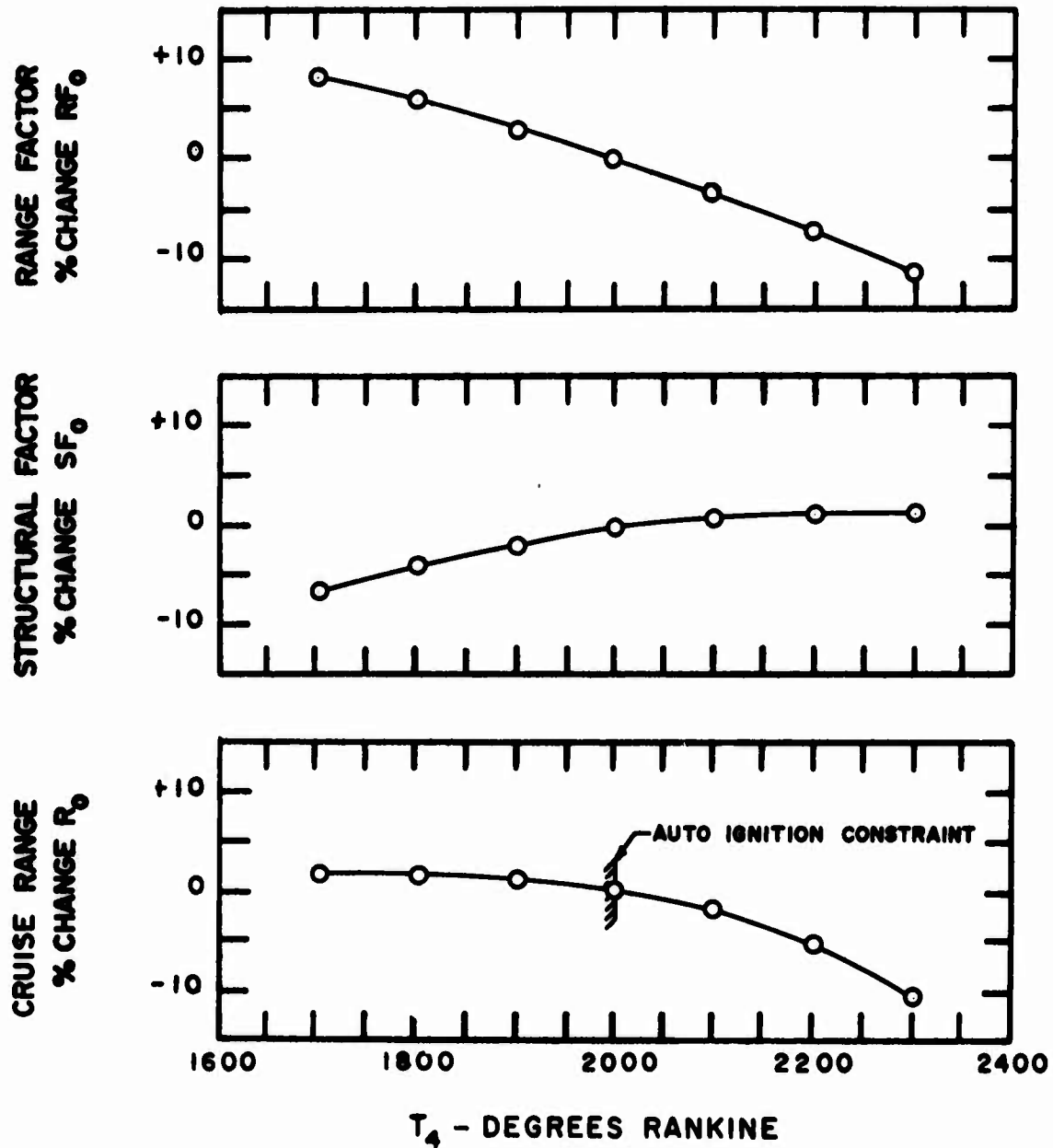


FIG 14 Combustor Entrance Temperature T_4 Influence on Cruise Range and Factors of Breguet Range Equation.

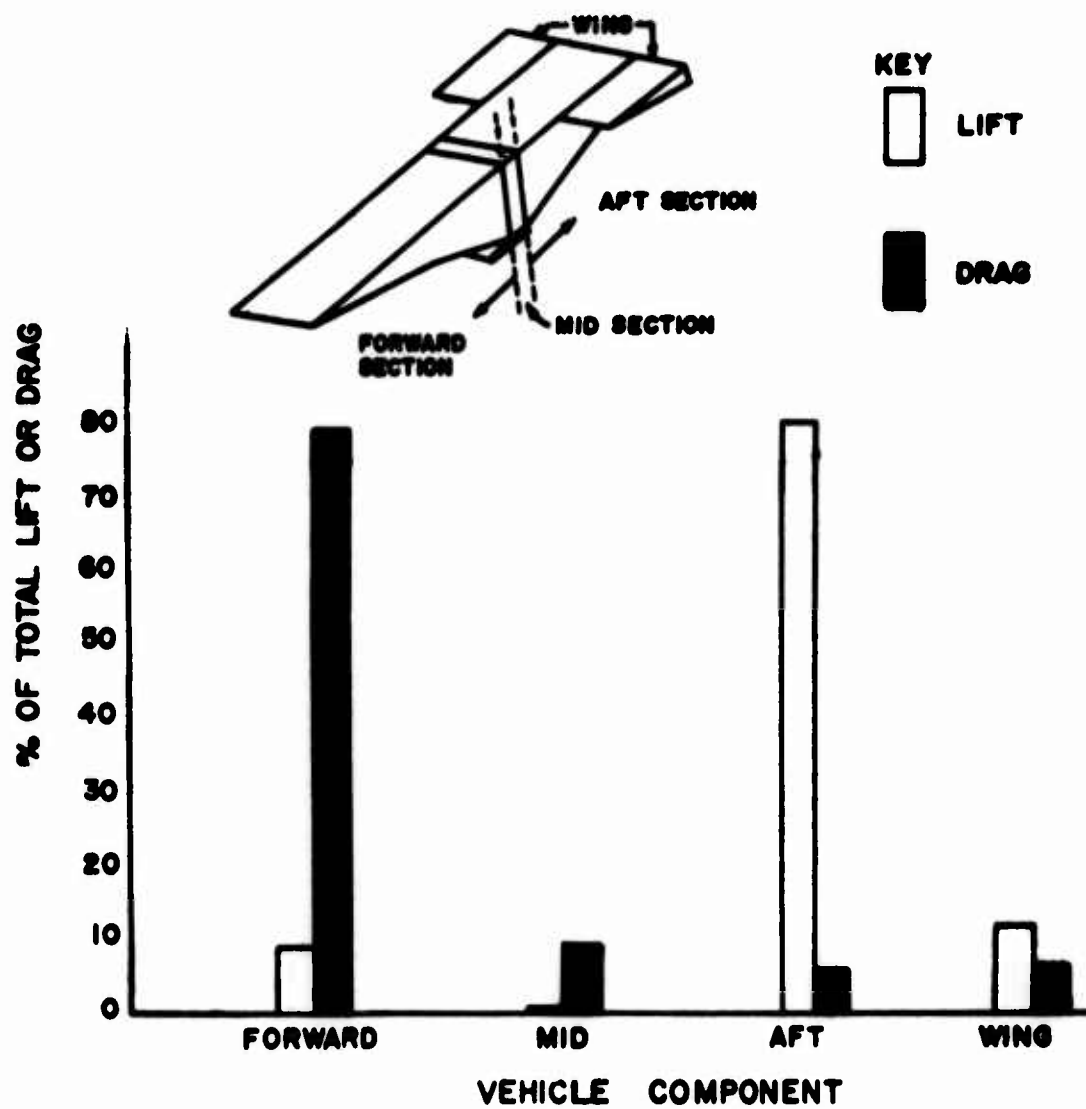


FIG 15 Vehicle Component Contribution to Lift and Drag of Mach 10 Configuration

may also be deduced from a simple momentum stream tube analysis, since for zero angle of attack and the shocks focused on the cowl lip, the inlet-forebody does not produce a net downward deflection of the inlet flow. A pitch up moment was produced by the forward section of the vehicle, however, which is reflected by the rearward placement of the wing on the configuration shown in Figures 10, 11, and 12.

An argument for the shock on lip at the design point constraint used in this study can be made using the stream tube analysis and some results of the study. When the shock pattern is focused ahead of the cowl lip, a downward deflection of the flow field occurs and the inlet-forebody becomes a wing in the sense that it produces lift, although with a highly compressed air flow. According to Kuchemann (Ref 3), it is not aerodynamically efficient to produce lift with a strong shock wave system. This fact was also evident from the range factor variation with lower wing surface angle δ_{11} shown in Figure 21. The aerodynamic efficiency, as reflected by the range factor, decreased with increasing shock strength since the strength of the shock wave produced by the wing increased with increasing lower wing surface angle. Thus, air which has been compressed by the inlet-forebody should be used in the propulsive stream (shock on lip) rather than to provide lift (shock ahead of lip) for optimum performance at the design point. Off design performance of the inlet was not included in this investigation; however, other studies (Ref 16, Ref 17) have shown the performance penalty to be small when fixed geometry inlets of the type considered in this investigation are operated below the design Mach number.

Combustor. Equivalence ratio ϕ and nondimensional combustor length l_4/h_4 are the independent design variables associated with the combustor midsection of the vehicle. As explained in Appendix E, the vehicle design problem was formulated such that the wing planform area was adjusted to generate drag to balance thrust produced at a given equivalence ratio. This is the inverse of the conventional

situation where vehicle geometry is fixed and the equivalence ratio is adjusted (by throttle setting) to maintain equilibrium flight during cruise. Hence, in the present vehicle design formulation, wing planform area S_p increased with increasing equivalence ratio ϕ . This is reflected by Fig 16 as the structural factor decreased with increasing equivalence ratio illustrating that wings were not volumetrically efficient compared to the fuselage of the vehicle.

As the planform area S_p of the wing decreased, the center of pressure of the wings moves rearward in order to maintain a trimmed condition. The trim constraint indicated in Fig 16 occurred when the wing had moved rearward to a position such that the common wing-fuselage length was zero ($l_b - l_{ew} = 0$). Although the optimum equivalence ratio occurred at the trim constraint, from a practical standpoint, some common wing-fuselage length ($l_b - l_{ew}$) would be necessary in order to attach the wings to the fuselage without resorting to booms or other such devices. Figure 16 illustrates that the performance penalty for increasing the common wing-fuselage length was slight. The configurations in Figs 10 - 12 are shown with a common wing-fuselage length of $(l_b - l_{ew})/l_1$ equal to three tenths. Figure 16 also illustrates that only a small performance gain could be obtained by relaxing the trim constraint, since the range for a wingless body ($S_p = 0$) was only slightly greater than for a configuration with sufficient wing to maintain trimmed flight.

The combustor length l_4 specifies the length of the combustor as well as the length of the midsection of the vehicle. Although the structural factor increased with increasing vehicle midsection length, the range factor decreased at an even faster rate due to skin friction drag in the combustor as shown in Figure 17. Since the chemical kinetics of mixing, ignition, and burning were not considered in the combustor synthesis (only the initial and final states of the fuel-air mixture are considered), nothing can be said about the optimum length of the combustor from this investigation other than that the combustor

$RF_0 = 11191$ NAUTICAL MILES
 $SF_0 = 0.4269$
 $R_0 = 4777$ NAUTICAL MILES

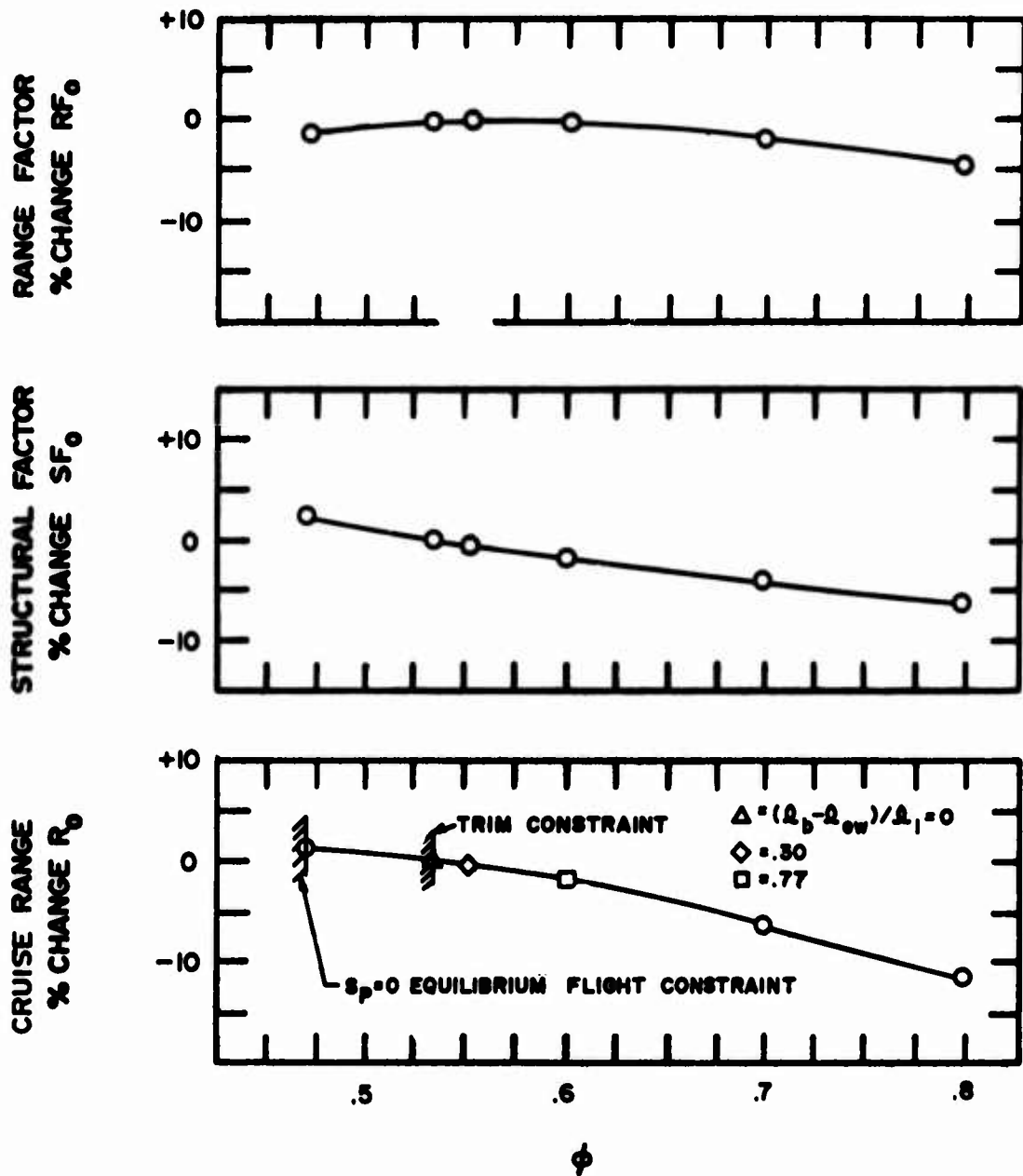


FIG 16 Equivalence Ratio ϕ Influence on Cruise Range and Factors of Breguet Range Equation

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4269$

$R_0 = 4777$ NAUTICAL MILES

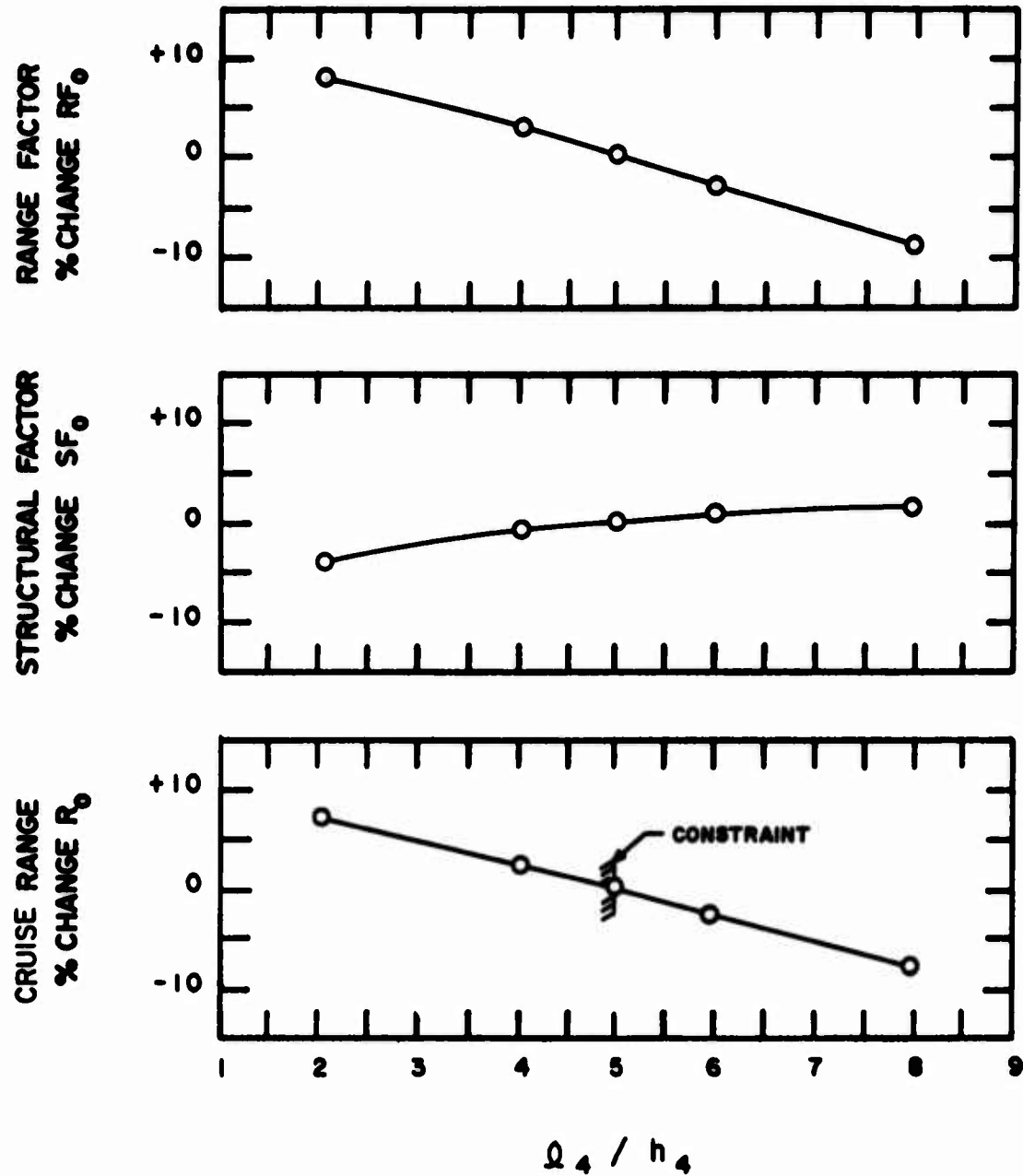


FIG 17 Nondimensional Combustor Length l_4 / h_4 Influence on Cruise Range and Factors of Breguet Range Equation

should be as short as possible. A combustor length to height ratio of $l_4 / h_4 = 5$, which was chosen as a lower constraint, resulted in a physical combustor length of $l_4 \sim 10$ ft for the three Mach number designs which provides a reasonable length for combustion to take place (Ref 18).

Nozzle-aftfuselage. Three independent design variables (δ_5, l_5, l_6) determine the geometry of the nozzle-aftfuselage of the configuration shown in Figures 10 - 12. The underside of the aft-fuselage provides an expansion surface for the combustor gases, thereby forming, along with the nozzle cowl and fences, a two-dimensional asymmetric nozzle. The nozzle can be configured so that the resultant nozzle force direction θ_T (θ_T measured from a plane perpendicular to the flight path) is in the direction of lift ($\theta_T = 0^\circ$) or in the direction of thrust ($\theta_T = 90^\circ$). Thus, the possibility of an optimum force direction θ_T exists.

Swithenbank (Ref 17) suggests the optimum nozzle-aftfuselage configuration for the two-dimensional asymmetrical nozzle would be obtained when the resultant nozzle force direction θ_T , in terms of the quantities defined in this investigation, is given by

$$\tan \theta_T = \left(\frac{L}{D_w} \right) \quad (11)$$

Appendix B contains the development of Eq 12 which is a more general expression than Eq 11 for the nozzle force direction angle θ_T producing a stationary value of the range factor:

$$\tan \theta_{Tst} = \frac{\left(\frac{L}{D_w} \right) + \frac{1}{C_T} \frac{\partial C_T}{\partial \theta_T}}{1 - \left(\frac{L}{D_w} \right) \frac{1}{C_T} \frac{\partial C_T}{\partial \theta_T}} \quad (12)$$

If the term $\partial C_T / \partial \theta_T$ which represents the change of nozzle force coefficient C_T with nozzle force direction θ_T is zero, as is usually the case for symmetrical nozzles, then Eq 12 reduces to Eq 11 the

result quoted by Swithenbank.

Equation 12 is essentially an expression indicating how the force recovered from the propulsive stream should be used to obtain optimum performance. If the wing is aerodynamically efficient (high L/D) and the thrust magnitude is independent of direction ($\partial C_T / \partial \theta_T = 0$), then the nozzle force direction should be nearly aligned with the vehicle flight path ($\theta_T \sim 90^\circ$). This is the case for most classes of subsonic and supersonic vehicles where the axis of the symmetrical engine nozzles are aligned with the reference axis of the vehicle. On the other hand, if the wings are inefficient converters of thrust to lift (low L/D), or if the loss in nozzle force with increasing θ_T is large, then Eq 12 indicates the nozzle configuration should be such that some lift is produced directly from the nozzle. This was the case for the class of vehicle configurations considered in this study, since Fig 15 illustrates that about 75% of the lift was produced directly by the nozzle-aftfuselage of the Mach 10 configuration.

In terms of vehicle geometry, a nozzle-aftfuselage designed to produce a force direction angle indicated by Eq 11 ($\theta_T = 76^\circ$ for $(L/D)_w = 4$) would have a large nozzle expansion half angle δ_s , and/ or a long nozzle cowl l_6 . For example, a nozzle force angle of $\theta_T = 76^\circ$ and a nozzle cowl length of $l_6 = 0$, implies a nozzle half angle of $\delta_s = 76^\circ$ for maximum range factor. Figure 18 illustrates maximum range factor was obtained at $\delta_s \sim 15^\circ$ for the Mach 10 configuration. The consideration of the structural factor shifted the optimum δ_s for maximum cruise range shown in Fig 18 to a slightly lower value than the δ_s for maximum range factor. At lower values of δ_s , more nozzle lift is produced allowing a smaller wing which improves the volumetric efficiency of the vehicle. Reference 19 reports that a survey of hypersonic cruise vehicle configurations, submitted by aircraft companies in response to a proposal request by the United States Air Force, revealed the nozzle half angle δ_s of the configurations to be

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4200$

$R_0 = 4777$ NAUTICAL MILES

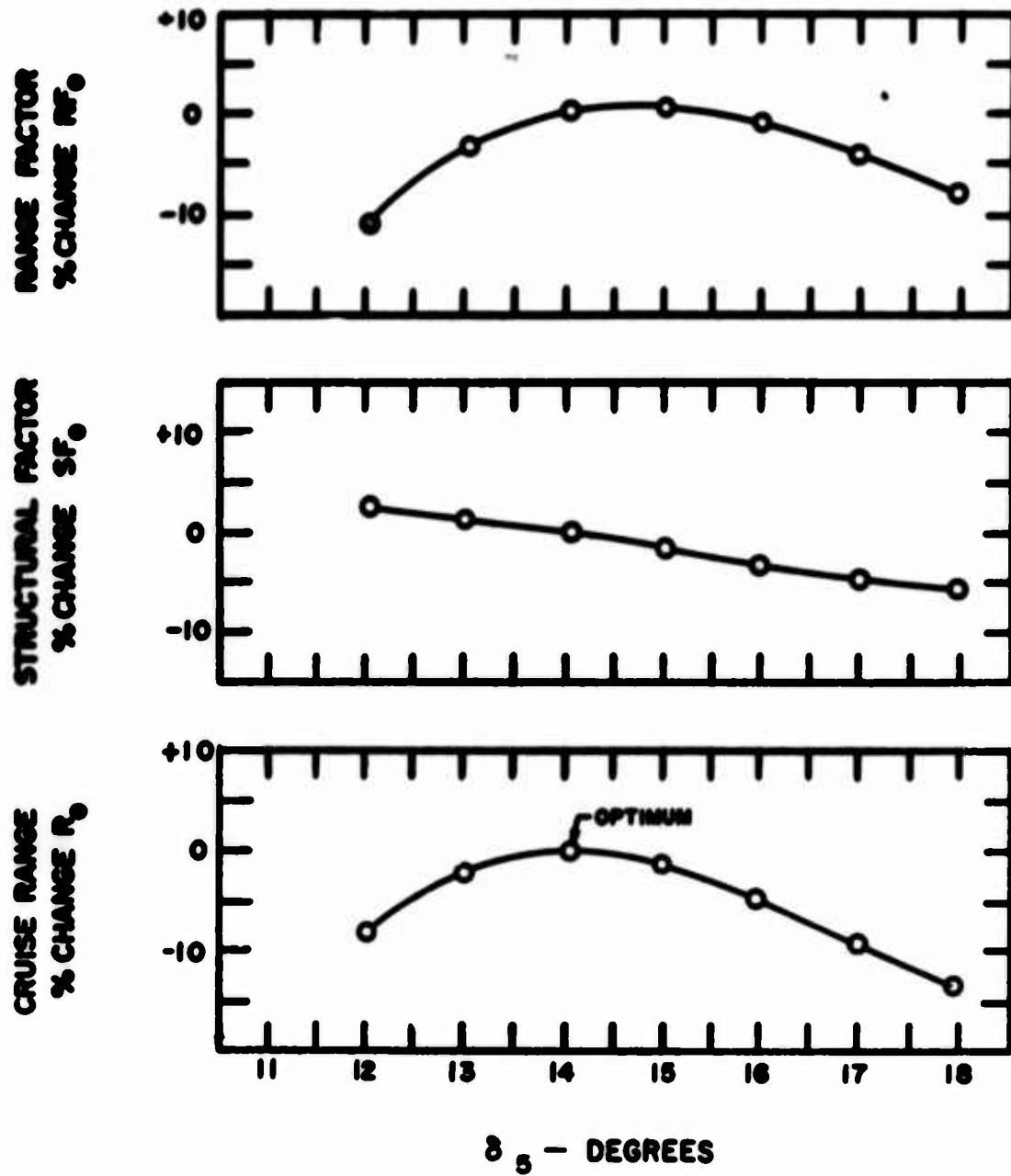


FIG 18 Nozzle Half Angle δ_s Influence on Cruise Range and Factors of Breguet Range Equation

approximately 15 degrees.

Although Fig 19 shows that nozzle cowl length l_c for maximum range factor occurred in the search domain, the decrease in structural factor with increasing cowl length caused the value of l_c for maximum range to occur at the minimum allowable length. As explained in Appendix E, the minimum nozzle cowl length was constrained such that the first down running characteristic or Mach line from the nozzle-aftfuselage expansion corner strikes the trailing edge of the cowl. However, Fig 19 illustrates low sensitivity of cruise range with nozzle cowl length in the region of the constraint, indicating the constraint value was at or near the optimum value.

The optimum nozzle length l_s occurred in the search interval for the Mach 8 design and at the zero base constraint for the Mach 10 and 12 designs. As shown in Fig 20, range factor increased with nozzle length, while the structural factor decreased. Thus, the sensitivity of range with nozzle length was low in the region of the optimum.

Wing. Optimum values of the independent configuration definition quantities associated with the wing (δ_{11} , δ_9 , δ_{12} , R) are shown in Table III. The insensitivity of cruise range with respect to the wing parameters, especially the parameters (δ_9 , R), accounts for the absence of a trend with design Mach number for optimum values of the wing parameters. Figure 15 illustrates a reason why performance was insensitive to values of the wing parameters as the wing lift is shown to constitute only about 10 per cent of the total vehicle lift. An inspection of Fig 21 reveals that the optimum lower wing surface angle δ_{11} occurred in the domain of the variable as a result of a trade off between the range factor and structural factor of the Breguet range equation. Figure 22 indicates that maximum range was obtained when the upper surface of the wing δ_9 was positioned to provide lift (negative incidence to the freestream flow) rather than to provide wing volume. Figure 23 illustrates that maximum range was obtained when

$RF_0 = 11191$ NAUTICAL MILES
 $SF_0 = 0.4269$
 $R_0 = 4777$ NAUTICAL MILES

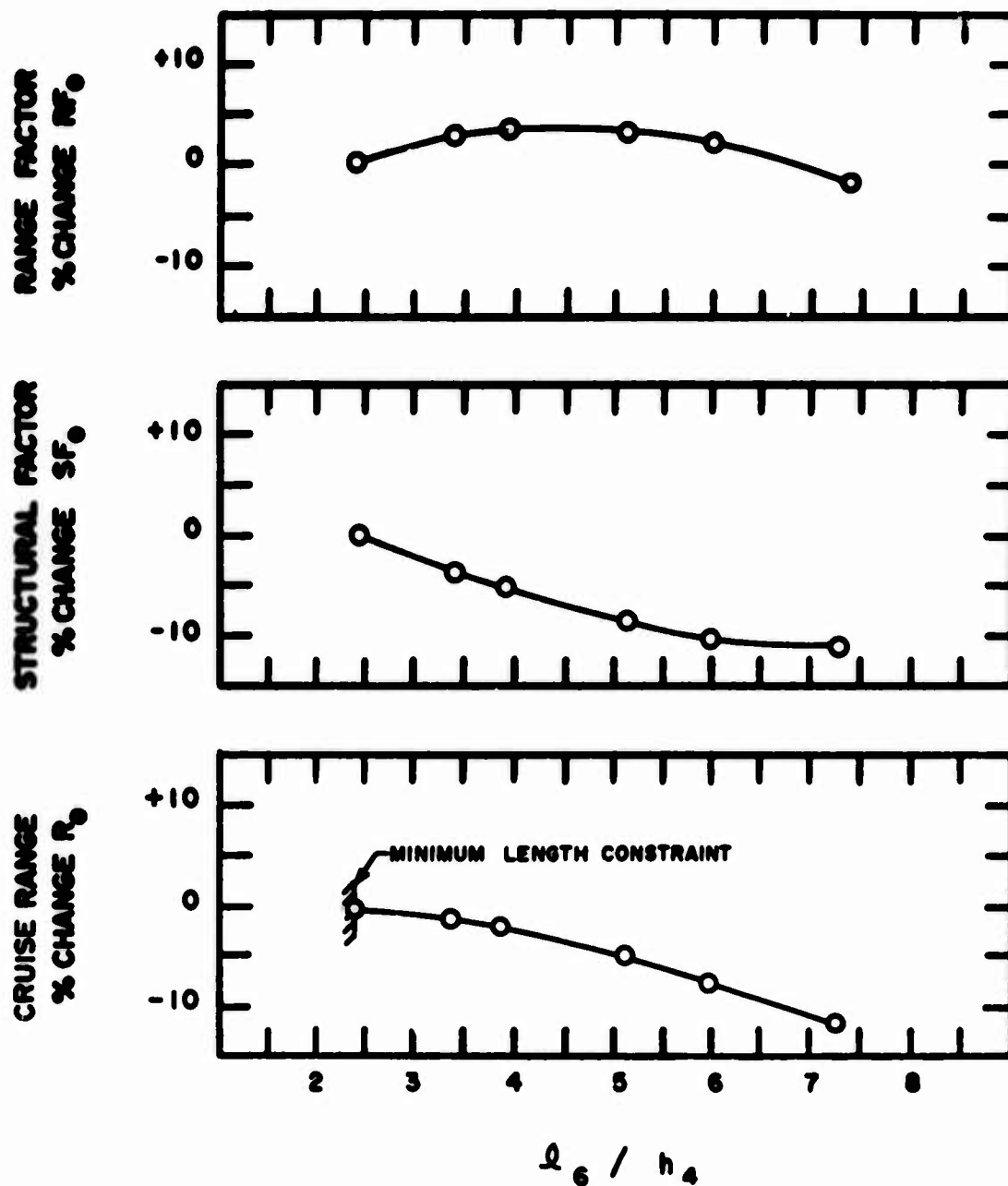


FIG 14 Nondimensional Nozzle Cowl Length l_6/h_4 Influence on Cruise Range and Factors of Breguet Range Equation

$RF_0 = 11191$ NAUTICAL MILES
 $SF_0 = 0.4269$
 $R_0 = 4777$ NAUTICAL MILES

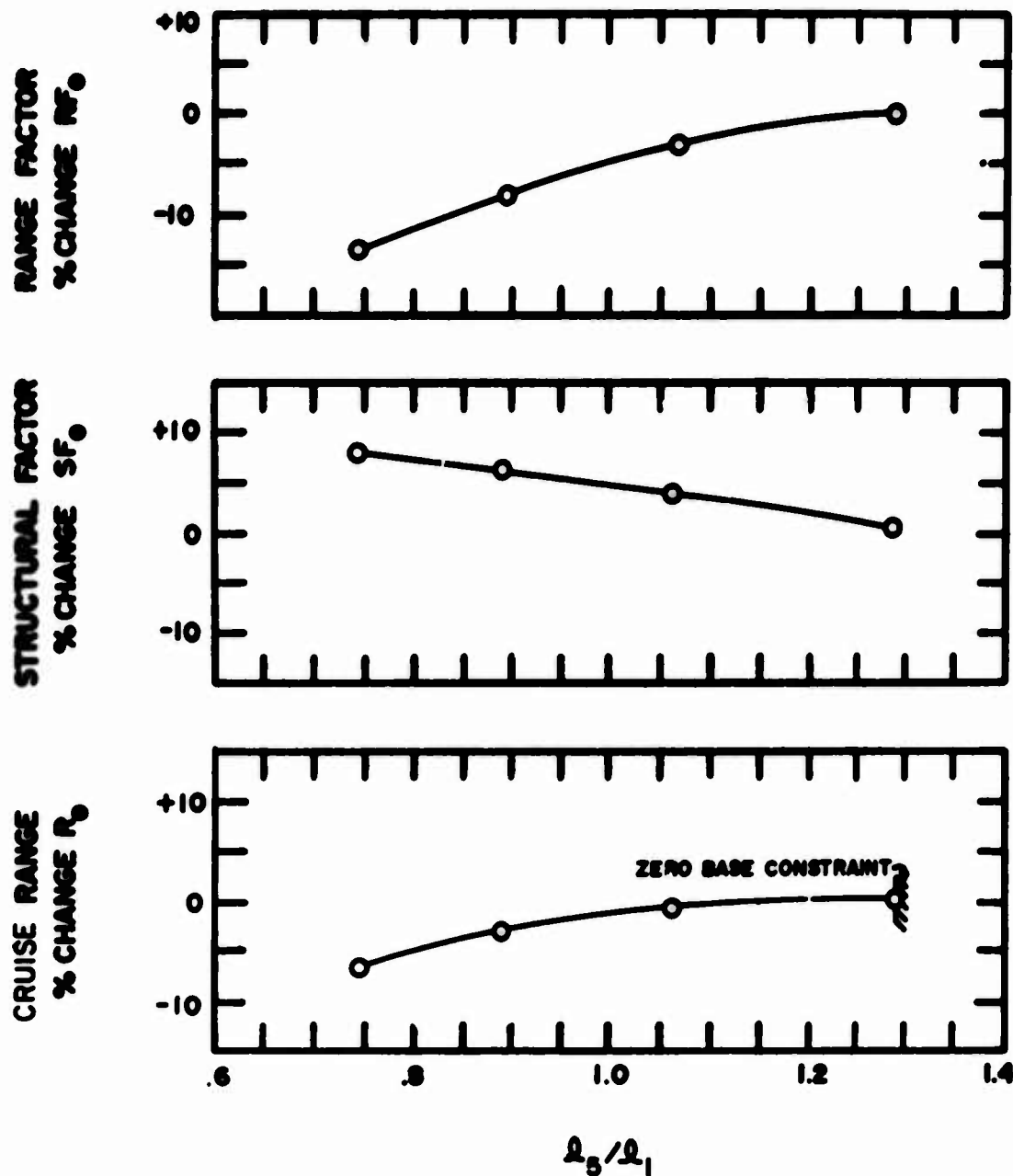


FIG 20 Nondimensional Nozzle Length l_2 / l_1 Influence on
 Cruise Range and Factors of Breguet Range Equation

$RF_0 = .1191$ NAUTICAL MILES
 $SF_0 = 0.4269$
 $R_0 = 4777$ NAUTICAL MILES

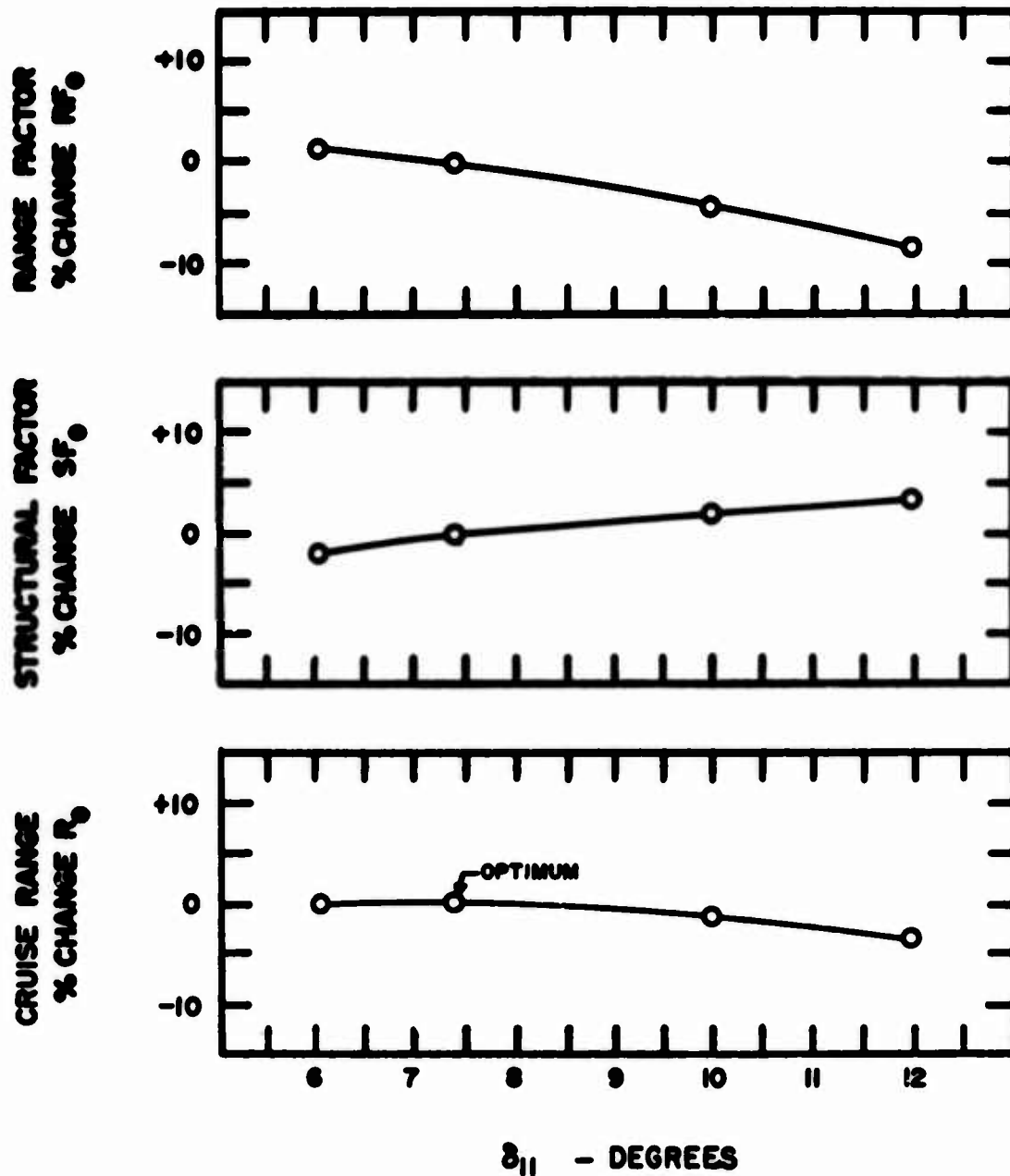


FIG 21 Lower Wing Surface Angle δ_{11} Influence on Cruise Range and Factors of Breguet Range Equation

$RF_0 = 11191$ NAUTICAL MILES
 $SF_0 = 0.4269$
 $R_0 = 4777$ NAUTICAL MILES

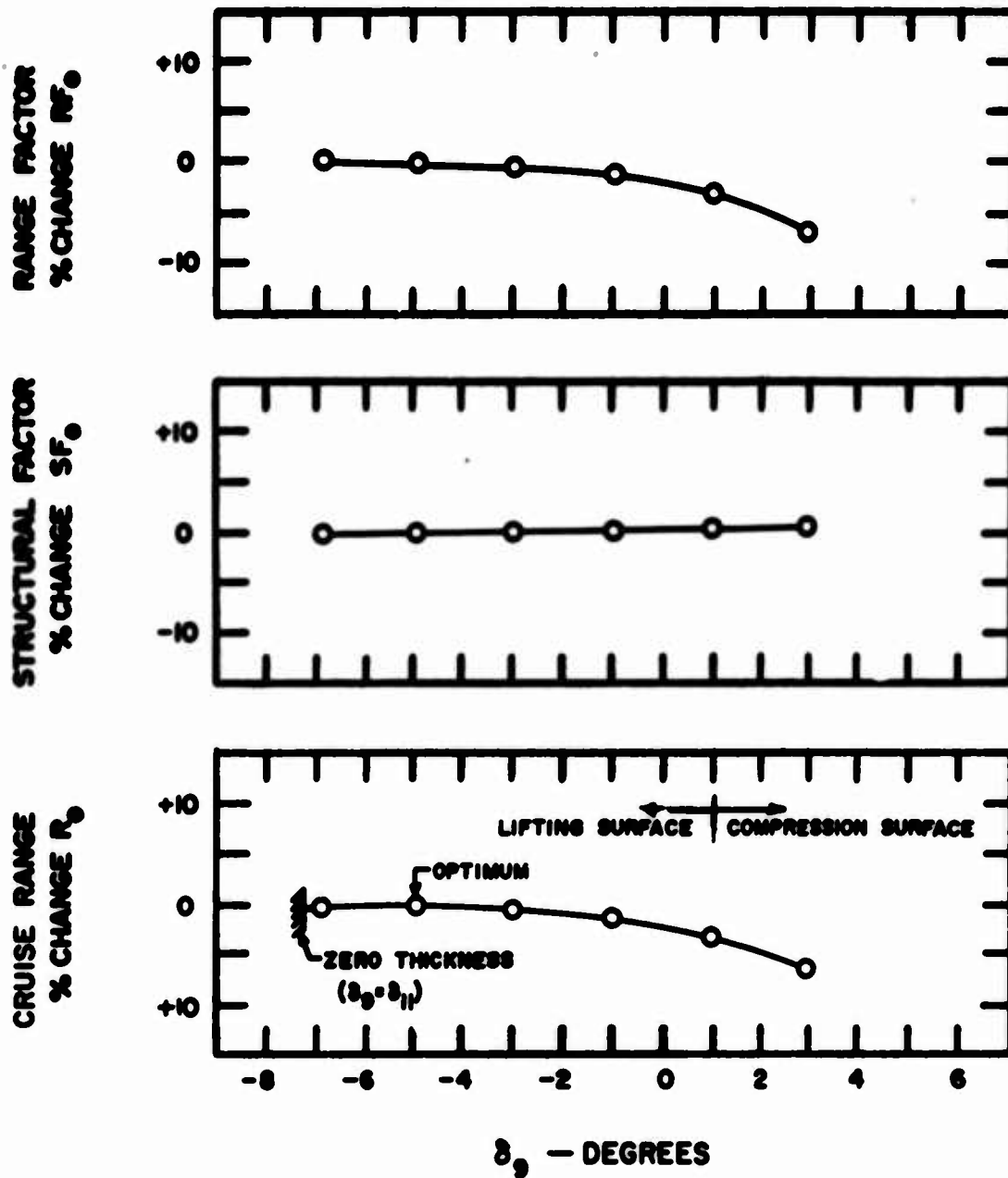


FIG 22 Upper Wing Surface Angle δ_u Influence on Cruise Range and Factors of Breguet Range Equation

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4209$

$R_0 = 4777$ NAUTICAL MILES

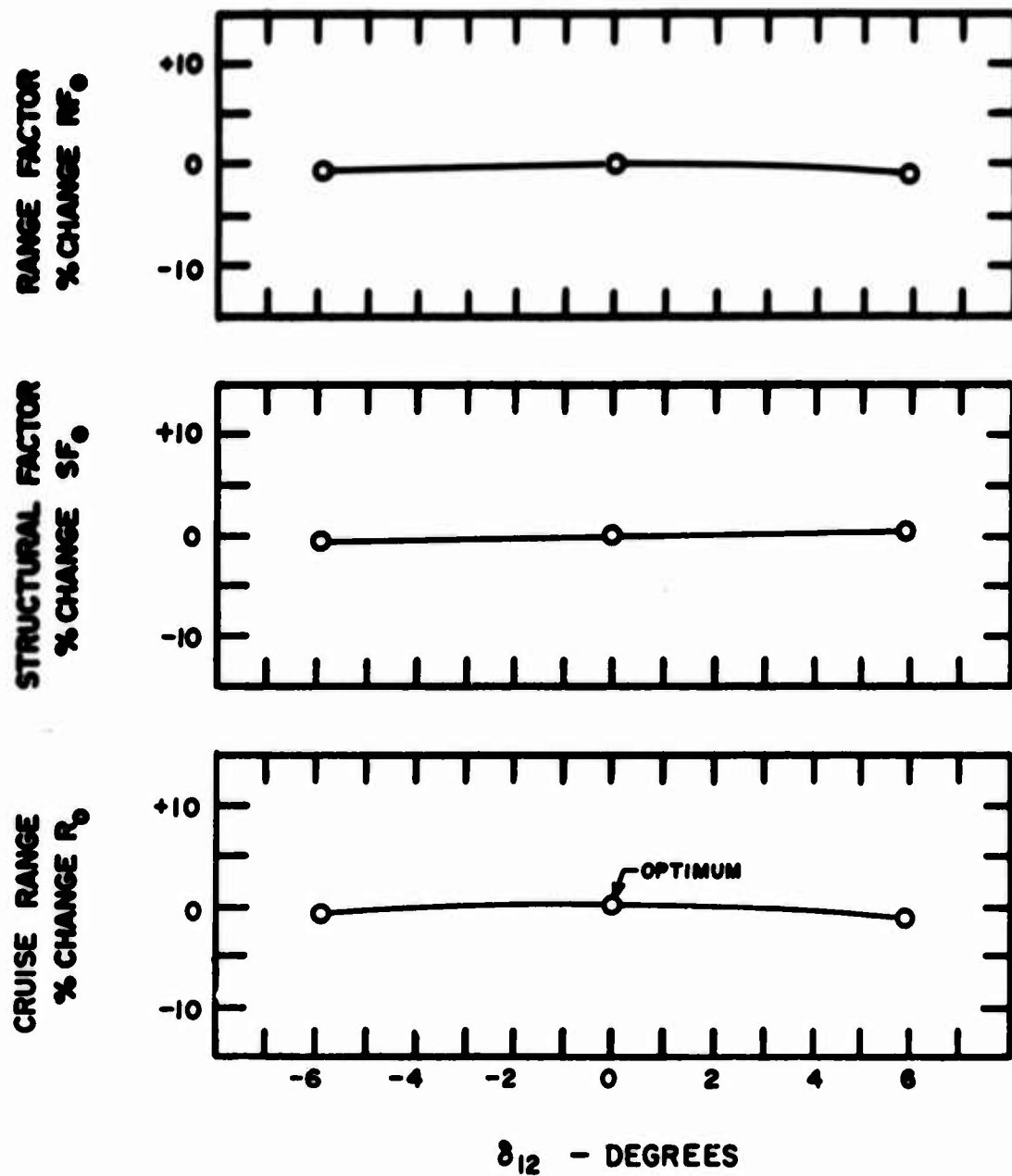


FIG 23 Wing Side Surfaces Angle δ_{12} Influence on Cruise Range and Factors of Breguet Range Equation

the side surface of the wing δ_{12} was at zero incidence with the free stream. Leading edge wing drag was a factor in the optimum wing being of low aspect ratio. However, if it were necessary to increase the wing aspect ratio due to takeoff and landing considerations, for example, Fig 24 shows that the performance penalty would be small.

Fuselage. Four independent design quantities are associated with the fuselage (α , δ_7 , W_{t0} , b/h). The optimum angle of attack α for the three design cases, as shown in Table III, was approximately one degree. A positive angle of attack permits the underside of the cowl to become a lifting surface. The lift provided by the cowl reduced the size of the wing which is reflected by the increasing structural factor with increasing angle of attack shown in Figure 25.

Although the upper fuselage surface angle δ_7 was constrained to be positive, a variable angle of attack allowed the upper fuselage surface to assume negative angles of incidence with respect to the free stream flow. Thus, the upper section of the fuselage could be positioned to provide lift or additional fuselage volume. Since Fig 26 was generated with an angle of attack of one degree, Fig 26 shows that more cruise range was obtained when the upper portion of the fuselage provided lift rather than volume, even though volumetric efficiency (as reflected by the structural factor) increased with increasing upper fuselage surface angle. Figure 26 also indicates maximum cruise range was obtained at the lower constraint boundary ($\delta_7 = 0$) as a result of the trade off between the range factor and structural factor.

The vehicle design takeoff weight W_{t0} for maximum cruise range occurred at the constraint boundary of 500,000 lb for all three Mach number designs. The predominant effect of variable takeoff weight is to change the dimensions of the vehicle through the scale factor l_1 of the vehicle. For a given geometrical shape the ratio of enclosed volume V to wetted area A_w increases with increasing

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4269$

$R_0 = 4777$ NAUTICAL MILES

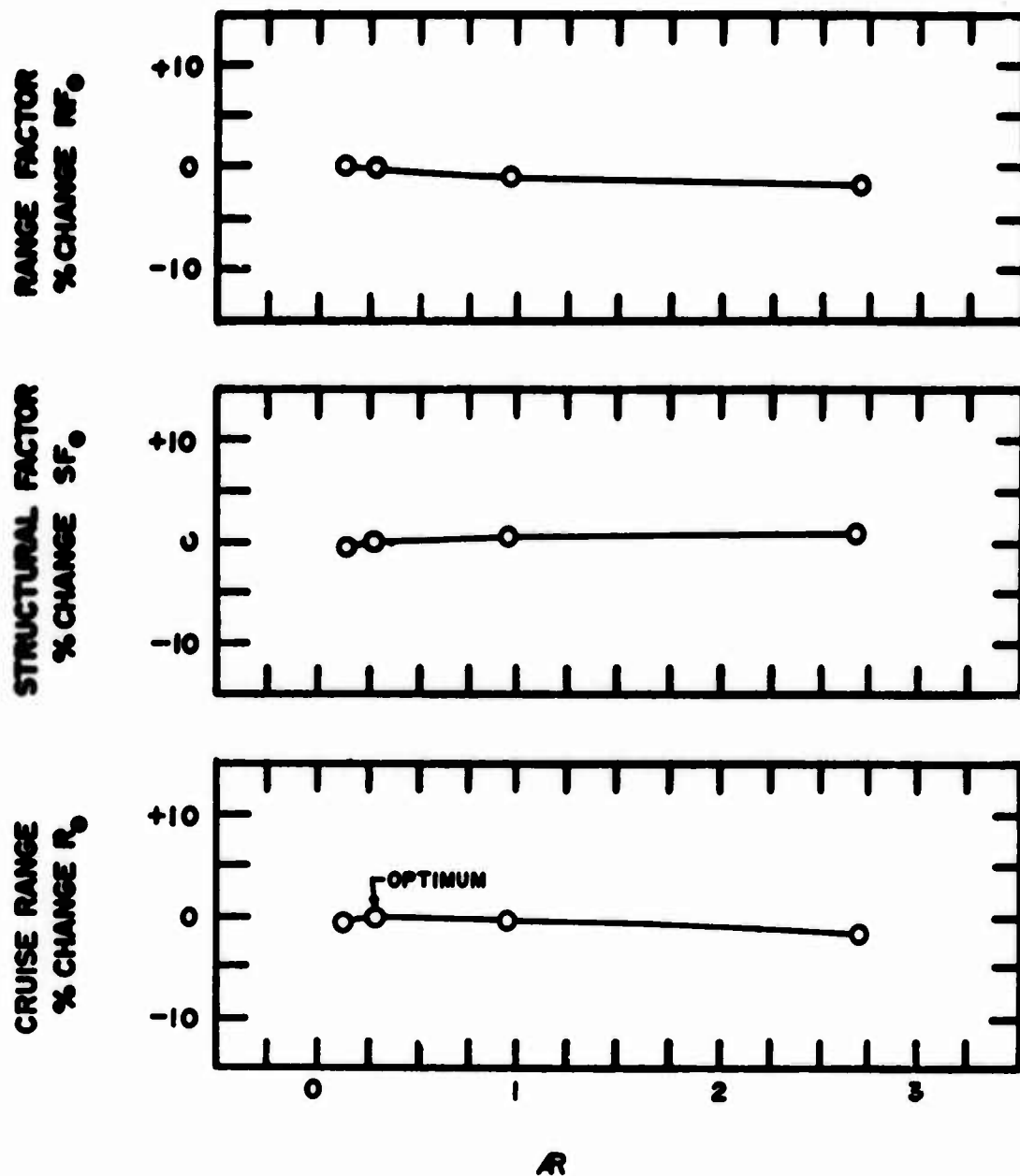


FIG 24 Aspect Ratio AR Influence on Cruise Range and Factors of Breguet Range Equation

$RF_0 = 11191$ NAUTICAL MILES
 $SF_0 = 0.4269$
 $R_0 = 4777$ NAUTICAL MILES

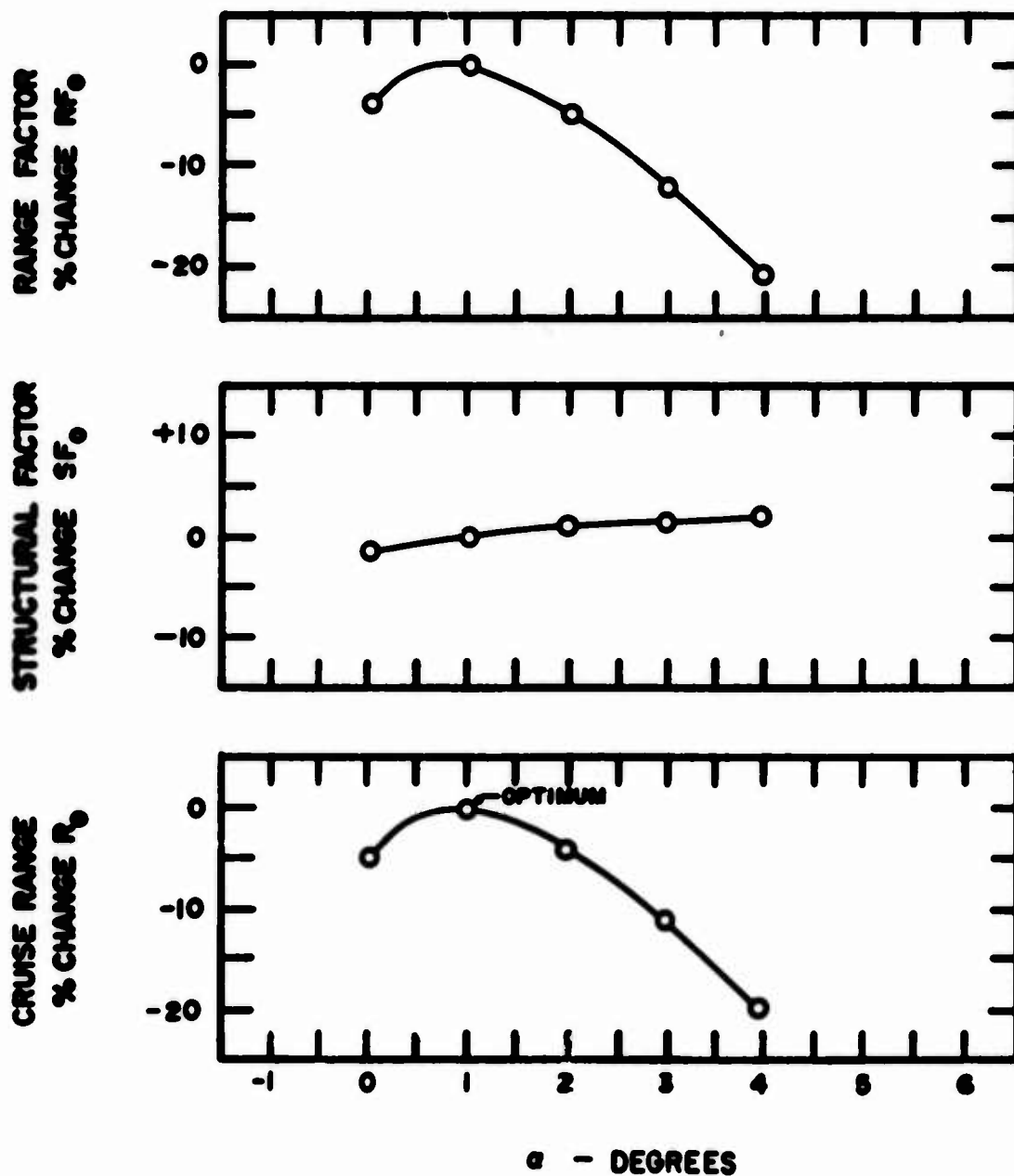


FIG 25 Angle of Attack α Influence on Cruise Range and Factors of Breguet Range Equation

$RF_0 = 11191$ NAUTICAL MILES
 $SF_0 = 0.4269$
 $R_0 = 4777$ NAUTICAL MILES

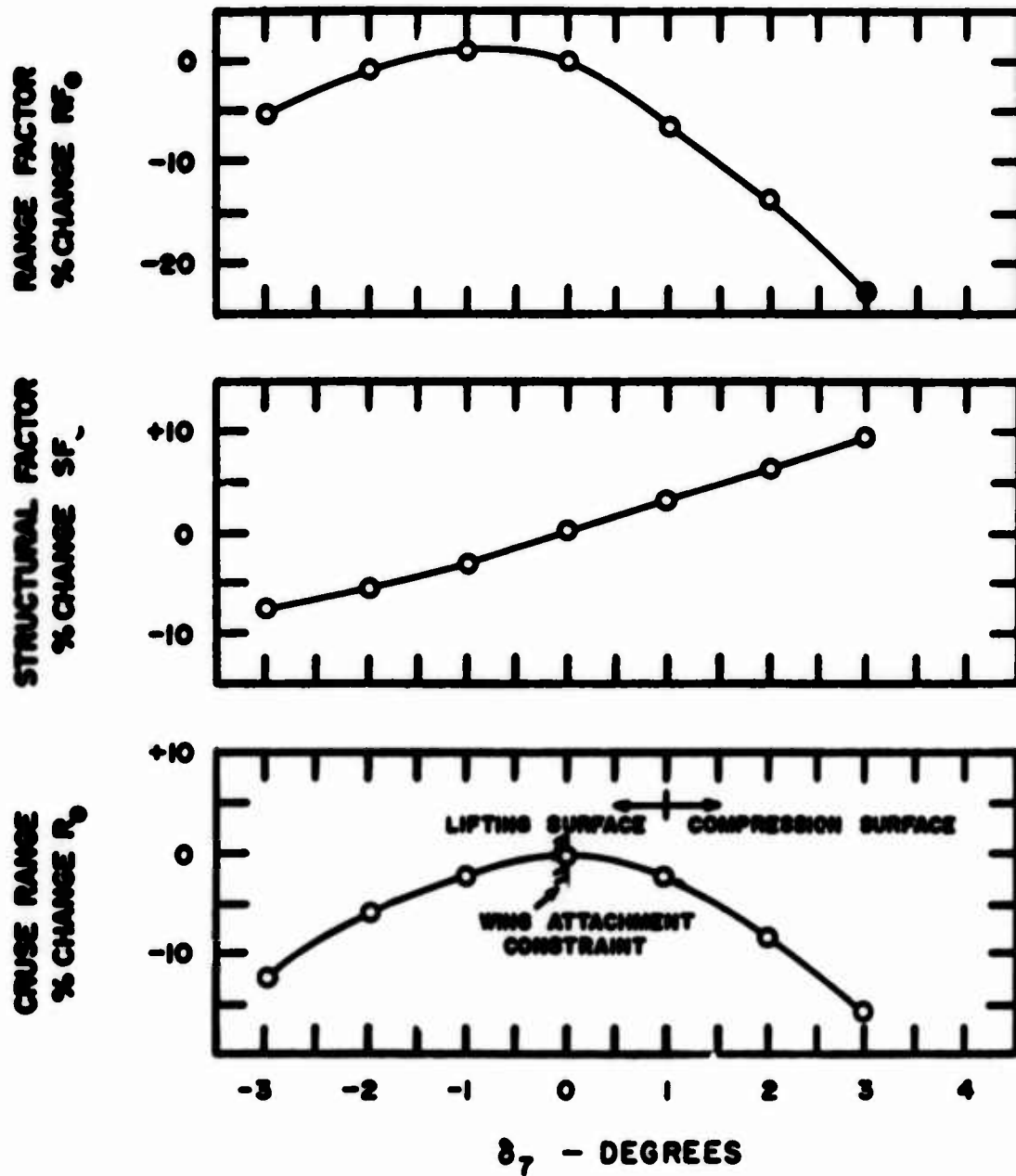


FIG 26 Upper Fuselage Surface Angle δ_7 Influence on Cruise Range and Factors of Breguet Range Equation

dimensions. The volume-wetted area ratio of a sphere, for example, increases in direct proportion to the radius. The increasing volume-wetted area ratio is reflected by the increase in structural factor with increase in design takeoff weight as shown in Fig 27 for the Mach 10 configuration. The trend of increased performance with increased size is evidenced in the subsonic cruise regime also by the appearance of larger jet aircraft such as the Boeing 747.

Width of the fuselage b was specified by the fuselage width to maximum thickness ratio b/h . The optimum value of the width to thickness ratio occurred as a result of a trade off between the range factor and structural factor as shown in Figure 28. Aerodynamic performance increased as vehicle width increased as a result of the friction drag of the vehicle sides (which is nearly constant) becoming a small percentage of the total drag as the width of the vehicle increased. Volumetric efficiency of the vehicle, as reflected by the structural factor in Fig 28, decreased with increasing fuselage width to thickness ratio after obtaining a maximum at width to thickness ratio of approximately one.

Range Potential

In addition to determining characteristics of optimum configurations in which lift, propulsion, and volume are integrated, a second objective of the study was to determine the possibility of range potential increase with increased design cruise Mach number. Range potential, as reflected by the range factor rather than absolute range, is treated here since the detailed weight and trajectory analysis necessary to determine the absolute level of range, was beyond the scope of this investigation. Figure 29 indicates for the cruise Mach number regime considered in this investigation (8 - 12) that the range potential increased with design cruise Mach number for the case without skin friction; however, for the case with skin friction, little change in range potential occurred with cruise Mach number.

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4269$

$R_0 = 4777$ NAUTICAL MILES

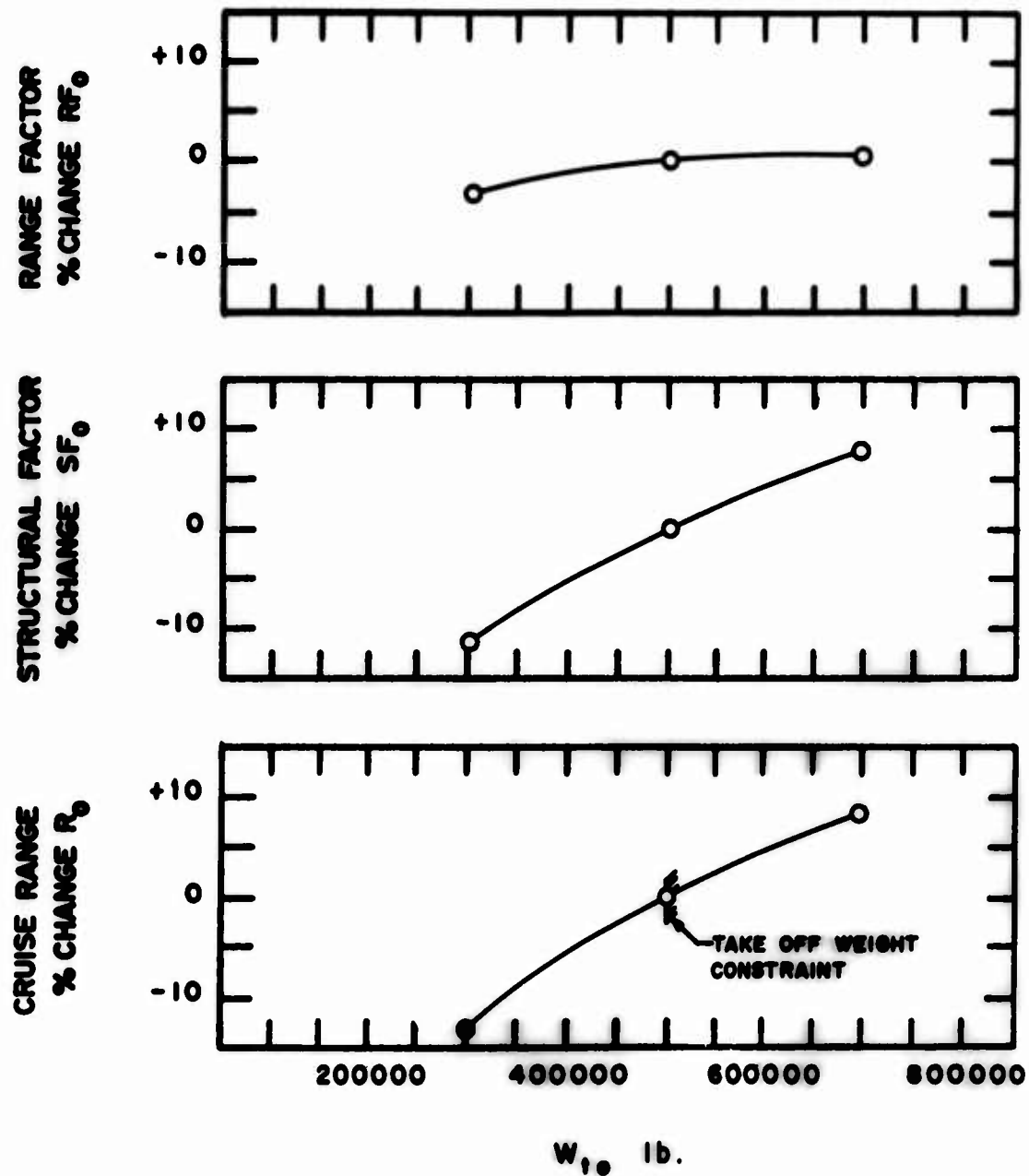


FIG 27 Take Off Weight W_{to} Influence on Cruise Range and Factors of Breguet Range Equation

$RF_0 = 11191$ NAUTICAL MILES

$SF_0 = 0.4269$

$R_0 = 4777$ NAUTICAL MILES

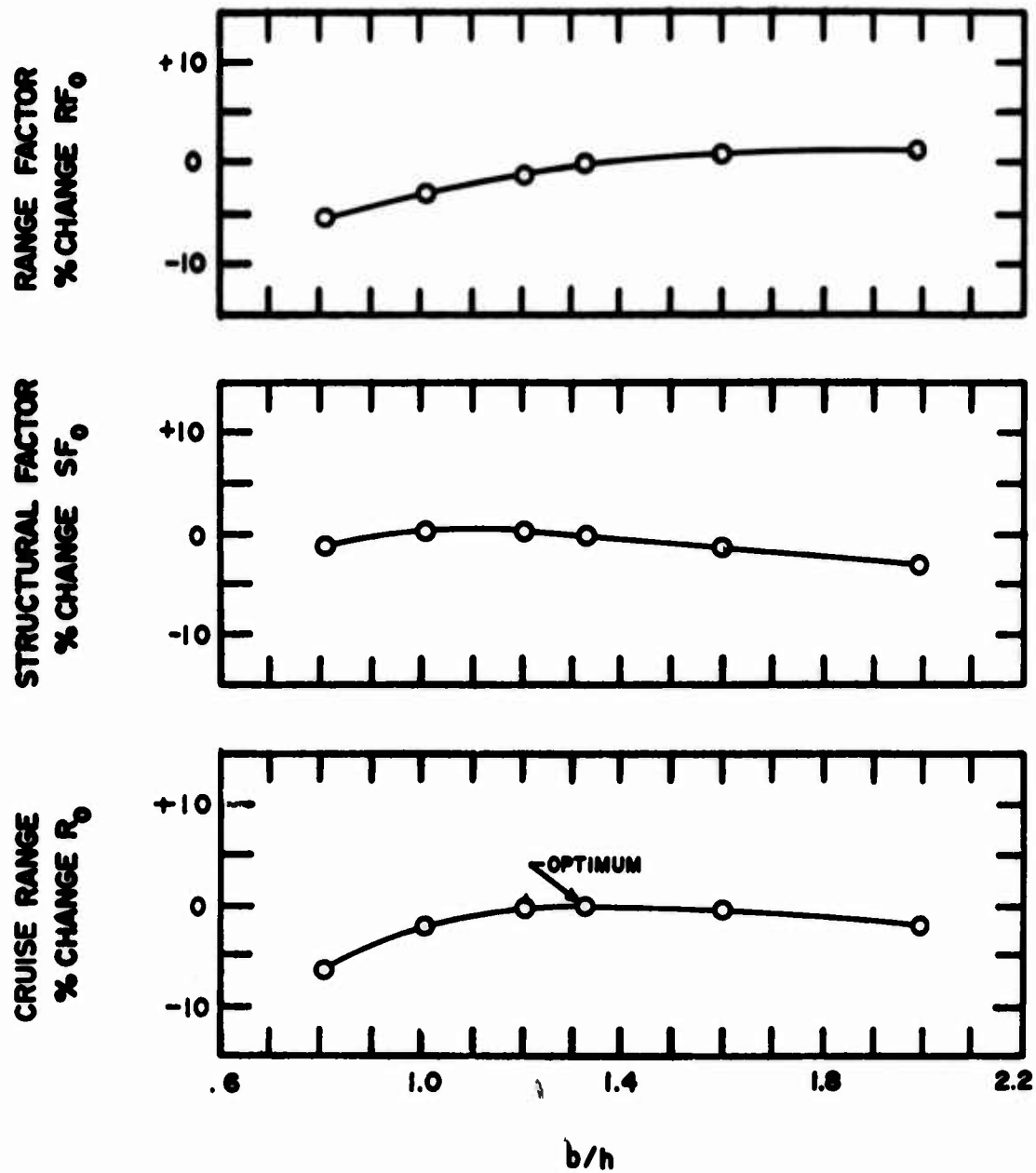


FIG 28 Nondimensional Vehicle Width b/h Influence on Cruise Range and Factors of Breguet Range Equation

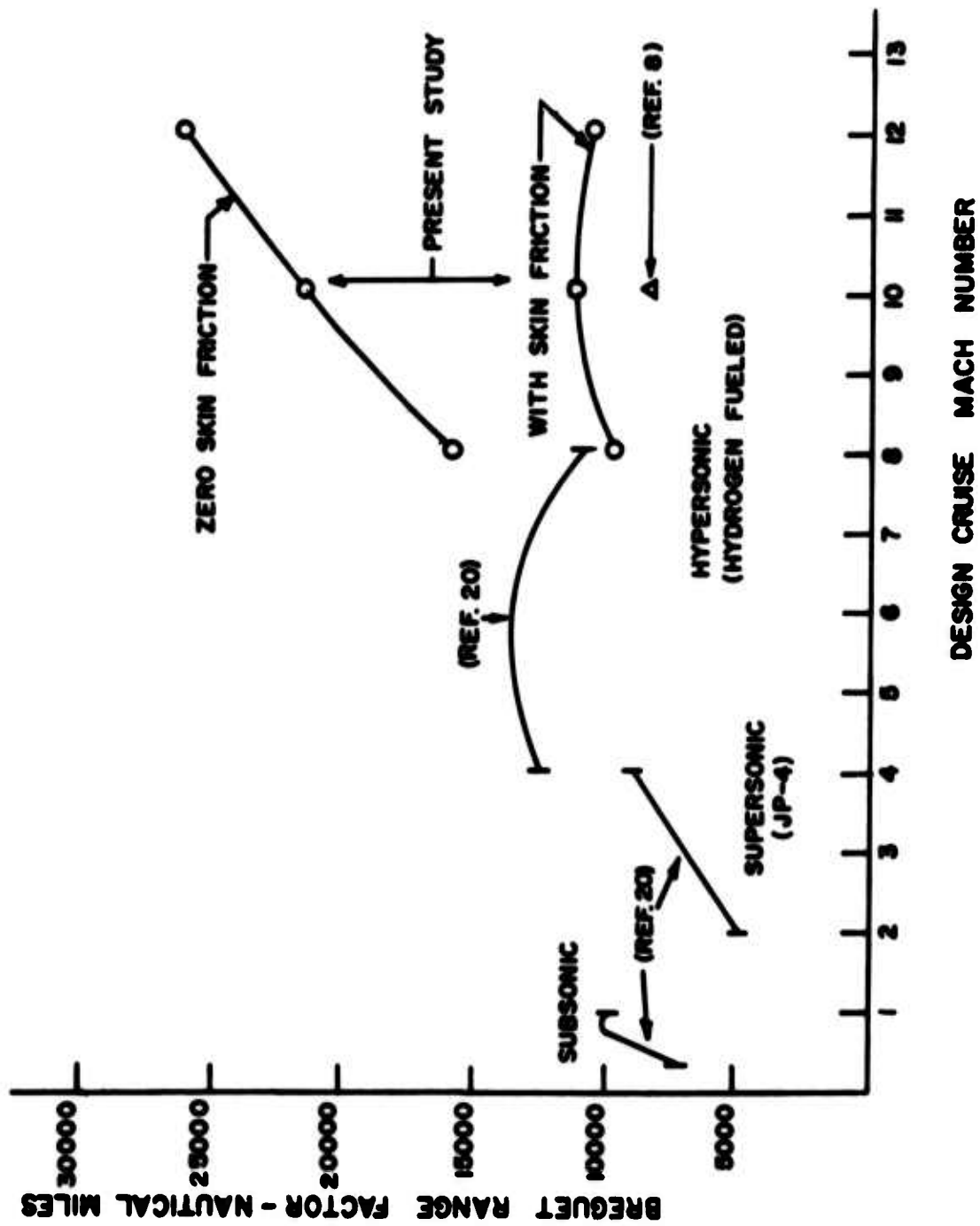


FIG 29 Range Potential of Transports

Figure 29 also indicates that skin friction places an important limitation on the range of vehicles in the Mach 8 to 12 cruise regime. A partial reason for this is illustrated in Fig 30 as friction drag is seen to become an increasingly higher percentage of total drag as the cruise Mach number increases since the pressure drag coefficient decreases. The decreasing pressure drag coefficient with Mach number is due to the decreasing thickness ratio of the vehicle since the deflection required to produce compression waves that will produce a combustor entrance temperature of $T_4 \geq 2000 \text{ R}$ decreases as design Mach number increases.

Another reason for the high sensitivity of hypersonic cruise vehicles to skin friction is the high dynamic pressure of the propulsive stream which comes into contact with certain components of the vehicle, especially in the final stage of compression. In the case of the Mach 10 design, for example, the dynamic pressure at the entrance of the combustor is approximately 12 times that of free stream.

The cooling constraint, fuel required for regenerative cooling of internal surfaces of the propulsion system and the leading edges of the vehicle be less than or equal to fuel required for propulsion ($\phi_c \leq \phi$), was not encountered as shown in Figs 10 - 12; therefore, the constraint did not affect the range potential of the optimum configurations. Appendix D contains a breakdown of the regenerative cooling requirements of the various components of the Mach 12 vehicle. Of the components considered, the combustor required the most active cooling while the leading edges of the vehicle required the least.

Comparison of Performance

The Breguet range factor is shown in Fig 29 for subsonic jet transports, supersonic transports, and a hypersonic (Mach 4 - 8) transport which was based on a performance study (Ref 20). Since the assumptions and conditions under which the values of Breguet range factor of these vehicles were obtained are not the same, the range

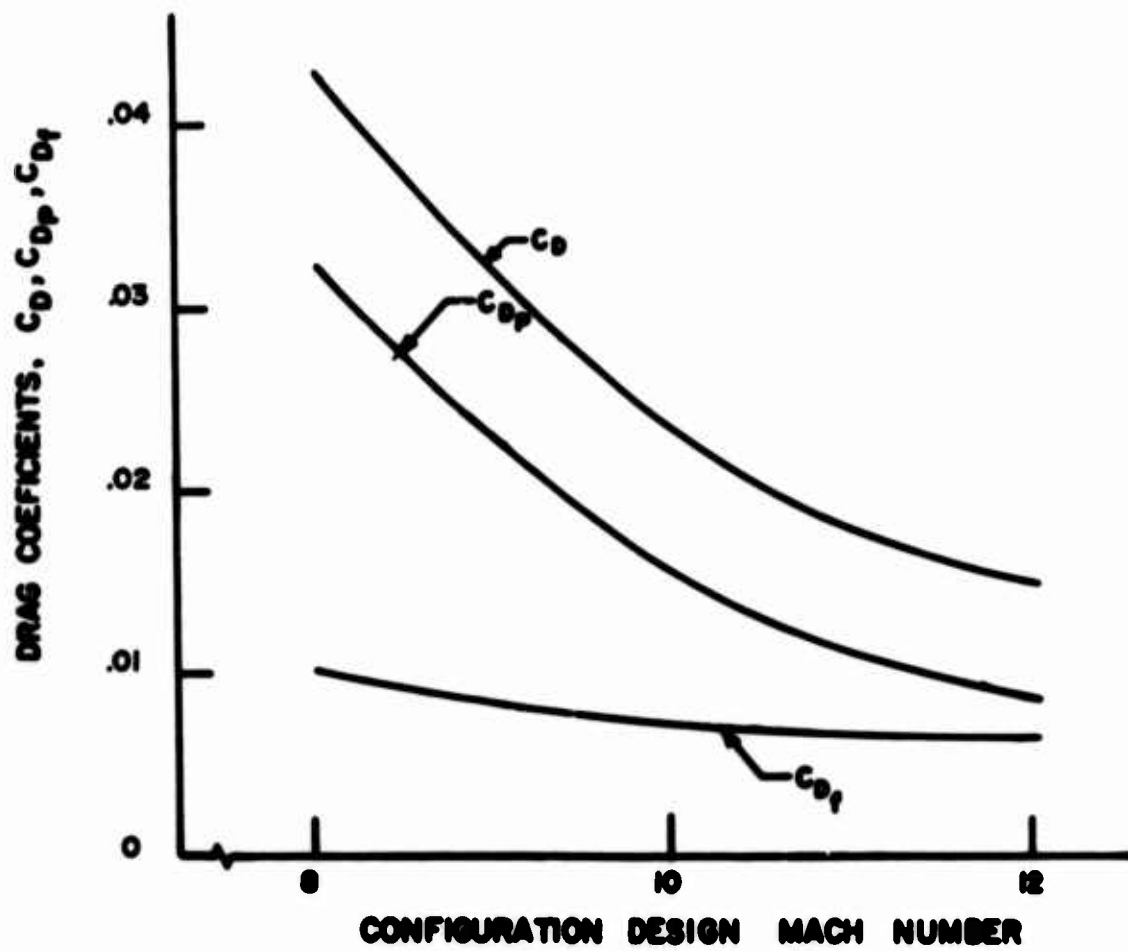


FIG 30 Variation of Drag Coefficients with Design Mach Number

potential of these vehicles is shown only to serve as a general guide to comparison of the cruise regimes. Figure 29 illustrates that the attractiveness of hypersonic cruise in the Mach 8 to 12 speed regime, from the standpoint of range potential increase over cruise below Mach 8, depends heavily on the level of skin friction drag. The lower the skin friction--the more attractive hypersonic cruise becomes.

Also included in Fig 29 is the range factor obtained by Williams (Ref 8) in a cruise vehicle optimization study. The study did not include the effects of skin friction; therefore, his result should be compared to the zero skin friction curve of the present study shown in Figure 29. The model for the optimization study by Williams contained three independent variables which were, in terms of the nomenclature used in this study, (δ_1 , δ_7 , δ_5). For the case of maximum range factor, which was the performance criterion for his study, the values of δ_1 and δ_7 obtained in both studies agree ($\delta_{1\text{ opt}} \approx \delta_1$ for maximum inlet pressure recovery, and $\delta_{7\text{ opt}} = 0$). However, his optimization model did not contain sufficient degrees of freedom to obtain higher performance which is a danger at any level of solution of optimization problems. If his model had included wings and or a variable equivalence ratio, the nozzle geometry would not have been constrained to produce equilibrium flight which resulted in the lower value of range factor shown in Figure 29.

Figure 31 contains a comparison of specific impulse I_{sp} data obtained in the present study and typical I_{sp} data obtained from one-dimensional supersonic combustion cycle analysis studies of hydrogen and air (Ref 17). Figure 31 illustrates that the specific impulse obtained from the present study decreases at a faster rate with increasing Mach number than the specific impulse obtained from the one-dimensional data. This is due to the fact that propulsive efficiency of the optimized configurations was traded for increased aerodynamic efficiency as design cruise Mach number increased. The objective of

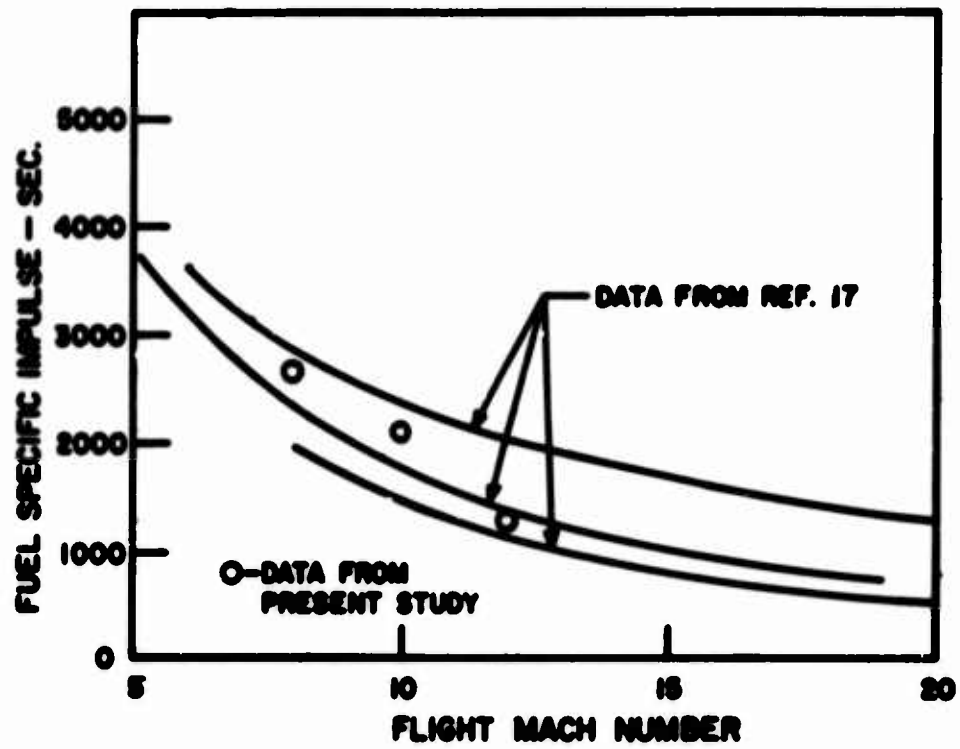


FIG 31 Specific Impulse Comparison

this investigation, however, was not to optimize a configuration based on the propulsive efficiency alone, but rather the geometry of a configuration in light of the combined propulsive, aerodynamic, and volumetric efficiencies. The main ideas of this chapter are summarized in the next chapter.

V. Summary and Conclusions

The optimization process was applied to the design of an air breathing hypersonic cruise vehicle configuration. Emphasis was placed on relating the configuration performance directly to the surrounding flow field so that any interaction between the aerodynamic and propulsive flow field would be accounted for in a fundamental manner. Cruise range, as determined from the Breguet range equation, was selected as the performance criterion for the optimization process with payload fraction becoming one of the constraints. The general class of vehicles which were optimized cruise in the Mach 8 to 12 speed regime, utilize a constant area supersonic combustion ramjet engine which burns hydrogen fuel, and can be characterized geometrically as two-dimensional wedgelike wing-body vehicles.

A generalized configuration model was defined by discrete parameters, transforming the variational optimization problem to a discrete or static optimization problem. Automated direct search algorithms were then used to determine the discrete parameters defining the configurations producing maximum cruise range for design cruise Mach numbers of 8, 10, and 12.

Within the limits of the analytical model used, the following qualitative conclusions can be deduced concerning the cruise configuration of the class of vehicles considered in this investigation:

- (1) The optimum configuration flies at about zero angle of attack.
- (2) Although developing very little lift, the inlet-forebody produces a pitch up moment.
- (3) Inlet-forebody compression ramp configuration for maximum range differs from the inlet-forebody configuration producing maximum pressure recovery (see Fig 13).
- (4) Inlet-forebody thickness ratio is determined by the minimum shock wave strength necessary to produce the autoignition air

temperature at the combustor entrance .

(5) Optimum combustor-midsection length of the vehicle is the minimum combustor length which allows for mixing, ignition and burning of the fuel.

(6) Optimum nozzle-aftfuselage configuration is one which produces a resultant nozzle force direction θ_T consistent with the expression derived in Appendix B:

$$\tan \theta_{T_{opt}} = \frac{\left(\frac{L}{D_w}\right) + \frac{\partial \ln C_T}{\partial \theta_T}}{1 - \left(\frac{L}{D_w}\right) \frac{\partial \ln C_T}{\partial \theta_T}}$$

(7) Maximum range is obtained when the upper surface of the fuselage is nearly parallel to the free stream flow.

(8) Range performance increases with takeoff weight which is consistent with the "cube-square law" preferring largest possible scale from a volumetric efficiency standpoint.

(9) Volumetric efficiency dictates a fuselage width slightly greater than the fuselage height (at the highest point) for maximum range.

(10) The wings of the optimum cruise configuration are small with approximately three-fourths of the total lift provided directly by the propulsive system.

In addition to the above conclusions concerning the cruise configuration of the vehicles, several qualitative conclusions can be deduced concerning performance trends:

(1) Range potential (Breguet range factor) increases with velocity as anticipated from the Breguet range equation for the case without skin friction. However; for the case with skin friction, little change in range occurs with velocity.

(2) Range factors of approximately 10000 nautical miles are indicated for transport type vehicles which cruise in the Mach 8 to 12 regime.

(3) The cooling capacity available in the liquid hydrogen fuel required for propulsion is adequate for cooling in the Mach range (8-12).

In regard to the direct method of configuration optimization, several factors can be noted.

(1) In this investigation, nongradient search algorithms were more efficient than gradient techniques such as the steepest ascent.

(2) Of the nongradient search algorithms, random point and sectioning were effective in the early phases of the search; whereas, the adaptive creeper search algorithm was effective in the terminal search phase.

(3) The performance response surface appeared unimodal in the region of the optimum and very flat so that a large number of configurations produced approximately the same performance.

Since the mathematical formulation of the problem did not include all classes of configuration or consideration of all constraints, influences, etc., the optimized configurations obtained in this investigation are not expected to offer a final solution to the optimum cruise vehicle problem. However, in addition to the fundamental configuration and performance results enumerated above, the investigation can contribute to a final solution by serving as a guide and as a comparison point for future configuration optimization studies in which the performance is related directly to the flow field.

Future work might include extension of the present model of the wedgelike wing-body class of vehicles to include subsonic combustion, or formulation of a new model to investigate another geometric family of shapes such as the conelike class of vehicles. In the more distant future however, as techniques progress for predicting the flow field about arbitrary three-dimensional shapes, as computers grow in size

and speed, as optimizing search algorithms become more efficient, the goal of configuration design will perhaps be realized. That is--when the configuration is shaped more by the physical aspects of the problem and less by the intuition of the designer.

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Appendix A

Cruise Trajectory Equations

The purpose of this appendix is to trace the development of a form of the Breguet range equation from the differential equations describing the state of the vehicle during the cruise portion of the mission profile.

With reference to Fig 32, the equations of motion of the vehicle (Ref 21) can be written for a nonrotating spherical earth in a wind axis coordinate system as

$$\frac{W}{g_c} \frac{dV_1}{dt} = T - D - W \sin \gamma \quad (A1)$$

$$\frac{W}{g_c} V_1 \frac{d\gamma}{dt} = L - W \cos \gamma + \frac{W}{g_c} \frac{V_1^2 \cos \gamma}{H + r_e} \quad (A2)$$

From Fig 32 the rate of change of altitude and ground range is given by

$$\frac{dH}{dt} = V_1 \sin \gamma \quad (A3)$$

$$\frac{dR}{dt} = \frac{r_e}{H + r_e} V_1 \cos \gamma \quad (A4)$$

The differential equation describing the rate of change of weight of the vehicle in terms of thrust and specific impulse is

$$\frac{dW}{dt} = -w_f = -\frac{T}{I_{sp}} \quad (A5)$$

Numerical methods are in general necessary to integrate Eqs A1 through A5 simultaneously from start to termination of cruise in order to determine the cruise range. If certain assumptions are made; however, the cruise range can be simply expressed as an ordinary function commonly referred to as the Breguet range equation. The

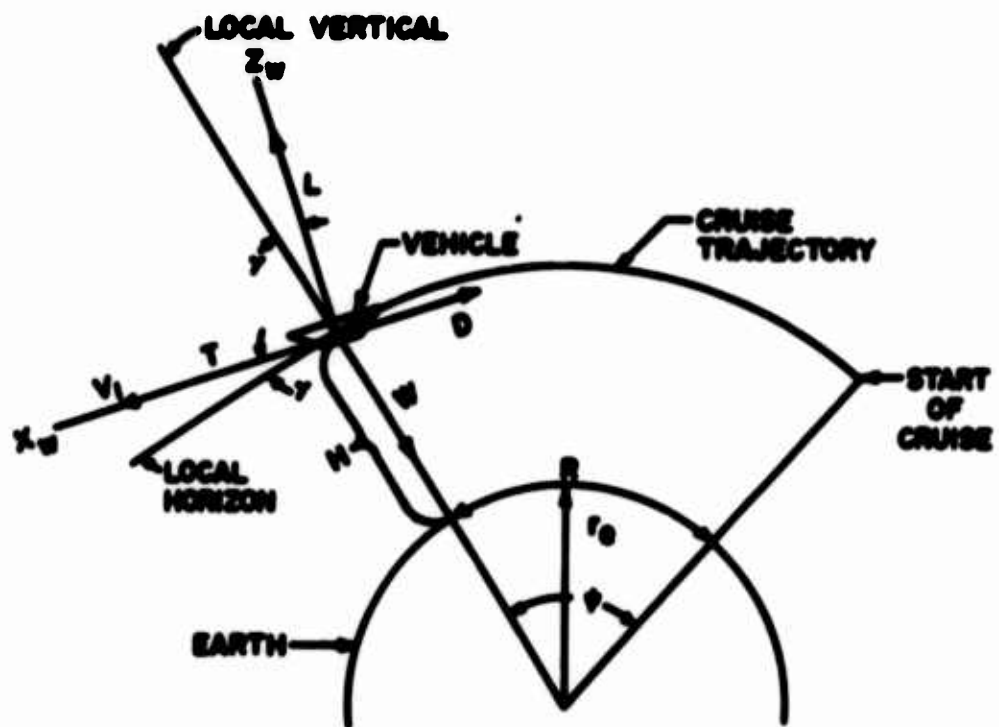


FIG 32 Cruise Trajectory Parameters

required assumptions are:

1. A zero flight path angle, $\gamma = 0$.
2. A constant flight path angle, $\frac{d\gamma}{dt} = 0$.
3. A constant cruise speed, $\frac{dV_1}{dt} = 0$.
4. A cruise altitude small compared to the radius of the earth, $\frac{r_e}{H + r_e} \approx 1$.

With the above assumptions Eqs A1 - A4 respectively become

$$T = D \quad (A6)$$

$$L = W \left(1 - \frac{V_1^2}{g_c r_e} \right) \quad (A7)$$

$$H = \text{constant} \quad (A8)$$

$$R = \int_{t_i}^{t_f} V_1 dt \quad (A9)$$

Equation A9 is integrated by changing the integration variable from time t to weight W

$$R = \int_{W_i}^{W_f} V_1 \frac{dW}{(dW/dt)} \quad (A10)$$

and by using Eqs A5 through A7 in Eq A10

$$R = \int_{W_i}^{W_f} - \left[\frac{L}{D} \text{Isp} \frac{V_1}{1 - \frac{V_1^2}{g_c r_e}} \right] \frac{dW}{W} \quad (A11)$$

The final form of the Breguet range equation is obtained by assuming that the bracketed term in Eq A11 is a constant for constant velocity cruise and noting that $g_c r_e = V_o^2$

$$R = \frac{L}{D} \text{Isp} \frac{V_1}{1 - V_1^2 / V_o^2} \ln \frac{W_i}{W_f} \quad (A12)$$

The form of the Breguet equation used as the performance criterion is obtained by substituting Eq A5 and A6 into Eq A12

$$R = \left[\frac{L}{\dot{W}_f} \frac{V_1}{1 - V_1^2 / V_o^2} \right] \ln \frac{W_i}{W_f} \quad (A13)$$

The range of each trial configuration during the optimization process was estimated with a range factor (the bracketed terms in Eq A11 and A13) evaluated at initiation of cruise. In order that the range factor remains constant for the cruise segment (necessary for the integration of Eq A11), the vehicle is assumed to climb during cruise as fuel is depleted (maintain $W/\rho_1 = \text{const.}$). A slight acceleration is induced, however, for a constant Mach number cruise as temperature of the standard atmosphere increases with altitude. For example, the initial and final cruise altitudes of the Mach 10 cruise vehicle were 115,000 ft and 124,000 ft respectively, which resulted in an ambient temperature increase of 14 R and an acceleration, to maintain constant Mach number cruise, of $dV_1 / dt = 1.88 \times 10^{-3} g$.

The change which occurs in flight path angle γ due to the altitude change is also slight. Thus, estimating the range at initial cruise and flying the constant Mach-varying altitude trajectory results in a reasonable cruise range prediction for the trial configurations evaluated during the optimization process.

Appendix B

Expression for Optimum Nozzle Thrust Angle

The purpose of Appendix B is to develop an expression for the nozzle thrust angle θ_T which produces a stationary or extremum value (if one exists) of the Breguet range factor.

Starting with the expression for the range factor

$$RF = \frac{L}{\dot{w}_f} \frac{V_1}{1 - V_1^2 / V_0^2} \quad (B1)$$

and noting that for a fixed cruise speed the only variable term in Eq B1 is L / \dot{w}_f which can be expressed as

$$\frac{L}{\dot{w}_f} = \frac{L_w + L_b + L_n}{(p_1 / R_1 T_1) A_c V_1 \phi f_s} \quad (B2)$$

where the wing lift L_w is given by

$$L_w = (F_n \sin \theta_T - D_b) \left(\frac{L}{D} \right)_w \quad (B3)$$

and the nozzle lift is given by

$$L_n = F_n \cos \theta_T \quad (B4)$$

Dividing numerator and denominator of Eq B2 by $l_1 b$ and nondimensionalizing the forces by $p_1 l_1 b$ results in the following expression

$$\frac{L}{\dot{w}_f} = \frac{(C_T \sin \theta_T - C_{D_b}) \left(\frac{L}{D} \right)_w + C_{L_b} + C_T \cos \theta_T}{(A_c V_1 \phi f_s) / (R_1 T_1 b l_1)} \quad (B5)$$

Applying the necessary conditions for an extremum with respect to the nozzle thrust angle to Eq B5, and noting that terms in the denominator of Eq B5 as well as C_{L_b} , C_{D_b} , and $(L/D)_w$ are independent of θ_T results in

$$\begin{aligned} \frac{\partial(L/\dot{w}_f)}{\partial\theta_T} &= \left(\frac{L}{D}\right)_w \left(C_T \cos \theta_T + \frac{\partial C_T}{\partial\theta_T} \sin \theta_T \right) \\ &+ \frac{\partial C_T}{\partial\theta_T} \cos \theta_T - C_T \sin \theta_T = 0 \end{aligned} \quad (B6)$$

Dividing Eq B6 by $\cos \theta_T$ and rearranging, produces an expression for the nozzle force direction angle producing a stationary value of the range factor:

$$\tan \theta_{T_{st}} = \frac{\left(\frac{L}{D}\right)_w + \frac{1}{C_T} \frac{\partial C_T}{\partial\theta_T}}{1 - \left(\frac{L}{D}\right)_w \frac{1}{C_T} \frac{\partial C_T}{\partial\theta_T}} = \frac{\left(\frac{L}{D}\right)_w + \frac{\partial \ln C_T}{\partial\theta_T}}{1 - \left(\frac{L}{D}\right)_w \frac{\partial \ln C_T}{\partial\theta_T}} \quad (B7)$$

Since the nozzle thrust direction θ_T is a function of the nozzle half angle δ_s , Eq B7 can also be used to determine the δ_s producing an extremum value of the range factor.

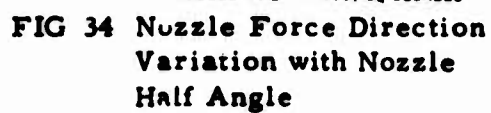
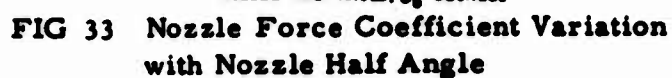
As an example, Figs 34 and 35 show the variation of nozzle force coefficient C_T and direction θ_T respectively as a function of nozzle half angle δ_s for a configuration of the class investigated in this study. In order to determine $\theta_{T_{st}}$ and $\delta_{s_{st}}$ for a configuration with a wing lift-drag ratio of $(L/D)_w = 5.5$ and an equivalence ratio of $\phi = .8$, a trial value of $\delta_{s_t} = 16^\circ$ is assumed. This allows the unknown term in Eq B7 to be evaluated graphically from Figs 34 and 35

$$\frac{1}{C_T} \frac{\partial C_T}{\partial\theta_T} = \left(\frac{1}{C_T} \frac{\partial C_T}{\partial\delta_s} \right) \left(\frac{\partial\delta_s}{\partial\theta_T} \right) = (-3.25) \left(\frac{1}{2.63} \right) = -1.27 \quad (B8)$$

substituting into Eq B7

$$\tan \theta_{T_{st}} = \frac{5.5 - 1.27}{1 + (5.5)(1.27)} = .530 \quad (B9)$$

which implies $\theta_{T_{st}} = 27.9^\circ$, and from Figure 35, $\delta_{s_c} = 15.8^\circ$.



This indicates the optimum nozzle half angle for maximum range factor predicted by Eq B7 is in the neighborhood of 16° which agrees reasonably well with the numerical solution of the range factor (Eq B1) as shown in Figure 36.

Appendix C

Performance Sensitivity with Respect to the Structural Set Parameters

In order to evaluate the structural factor (hence the cruise range) during the optimization of independent design variables, it was necessary to specify values of certain parameters. These parameters were termed the structural parameter set \bar{S} and consisted of

$$\bar{S} = \left[\rho_e, \rho_p, \rho_f, \frac{W_{fc}}{W_{fl}}, \frac{W_e}{W_{to}}, \frac{W_p}{W_{to}}, \gamma_s \right] \quad (C1)$$

Sensitivity of the cruise range and structural performance with each member of the set \bar{S} is shown in this appendix using the optimal Mach 10 design as a source for the value of the range factor (RF = 11191 nm) and geometrical shape quantities.

The geometrical shape provides a constant value of the nondimensional volume ($\Psi / l_1^3 = .2367$) and surface area ($A_w / l_1^2 = 4.568$). For a fixed takeoff weight ($W_{to} = 500,000$ lb) Eq E37 (Appendix E) becomes a cubic equation in the unknown scale factor l_1

$$\left(\frac{\Psi}{l_1^3} \right) l_1^3 + \left(\frac{\gamma_s}{\rho_f} \frac{A_w}{l_1^2} \right) l_1^2 - W_{to} \left[\frac{1}{\rho_f} \left(1 - \frac{W_p}{W_{to}} - \frac{W_e}{W_{to}} \right) + \frac{W_p}{W_{to}} \frac{1}{\rho_p} + \frac{W_e}{W_{to}} \frac{1}{\rho_e} \right] = 0 \quad (C2)$$

Once the scale factor l_1 is known, the structural weight is determined from

$$W_s = \left(\frac{A_w}{l_1^2} \right) \gamma_s l_1^2 \quad (C3)$$

and the fuel weight from

$$W_{fl} = W_{to} - W_p - W_e - W_s \quad (C4)$$

The structural factor of the Breguet range equation SF can now be evaluated from

$$SF = \ln \left[1 + \frac{\frac{W_{fc}}{W_{fl}} \frac{W_{fl}}{W_{to}}}{\frac{W_p}{W_{to}} + \frac{W_e}{W_{to}} + \frac{W_s}{W_{to}}} \right] \quad (C5)$$

Thus, the cruise range R can be determined for a constant range factor RF, since

$$R = (RF) (SF) \quad (C6)$$

Figures 36 through 43 were generated by varying the members of the structural parameter set one at a time while fixing the remaining members at the values used in the optimization study as shown in Table III. Since the range factor is constant, the percentage change in cruise range and structural factor is shown on the same ordinate in the figures. Figure 36 shows that the range was insensitive to wide variations of the equipment weight density. The performance is also shown in Fig 37 to be insensitive to payload density for perturbations about $\rho_p = 4.5 \text{ lb/ft}^3$ (density of airliner passenger compartments with people aboard).

Although the density of the liquid hydrogen fuel is fixed, Fig 38 is included to show, from a structural standpoint, how the performance increases as the density of hypothetical fuels increases. The fuels would have the same energy content per pound as hydrogen but with different densities. As a means of comparison, jet fuel (JP-4) has a density of approximately 50 lb/ft^3 ; whereas, the density of liquid hydrogen is 4.5 lb/ft^3 . Unfortunately, the energy content and heat sink capacity of JP-4 is very much lower than liquid hydrogen so that JP-4 is not considered an attractive fuel for the class of vehicles considered in this study.

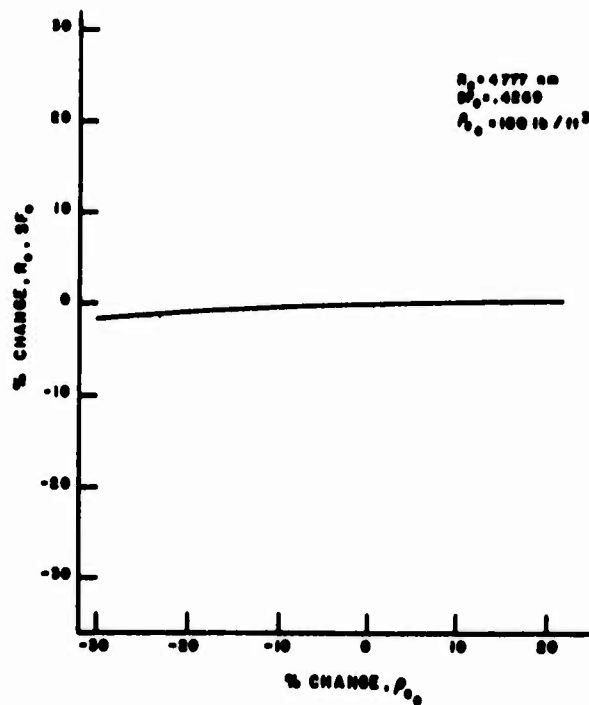


FIG 36 Equipment Density Influence on Performance

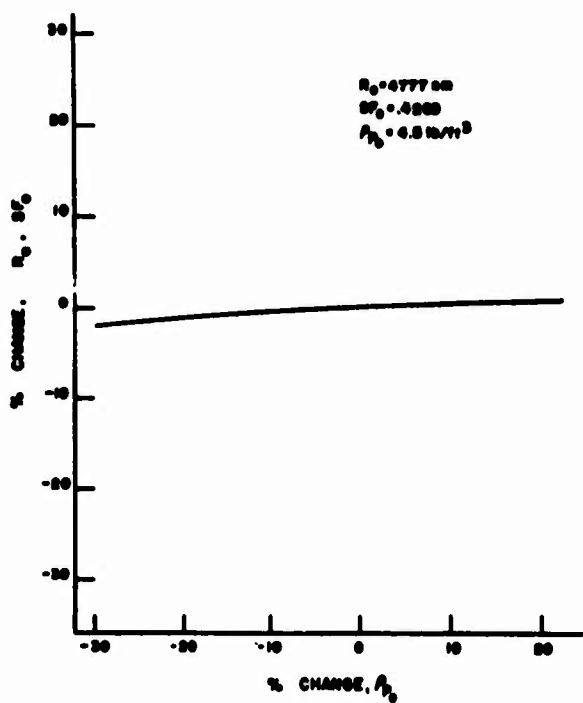


FIG 37 Payload Density Influence on Performance

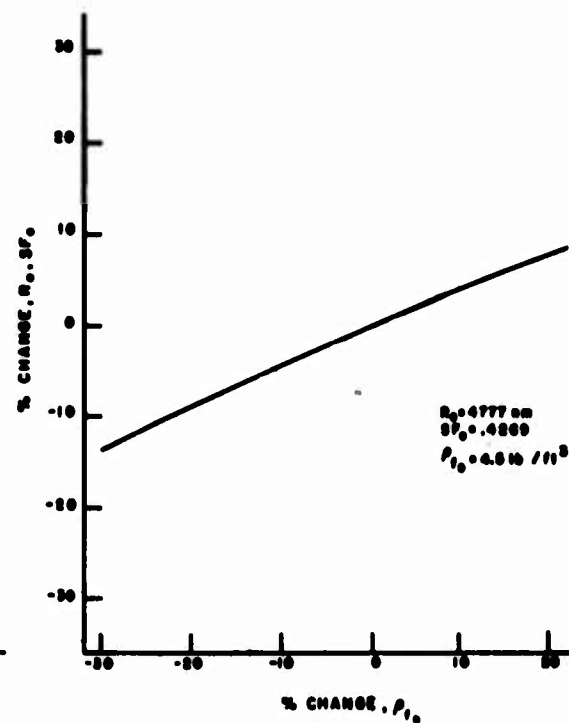


FIG 38 Fuel Density Influence on Performance

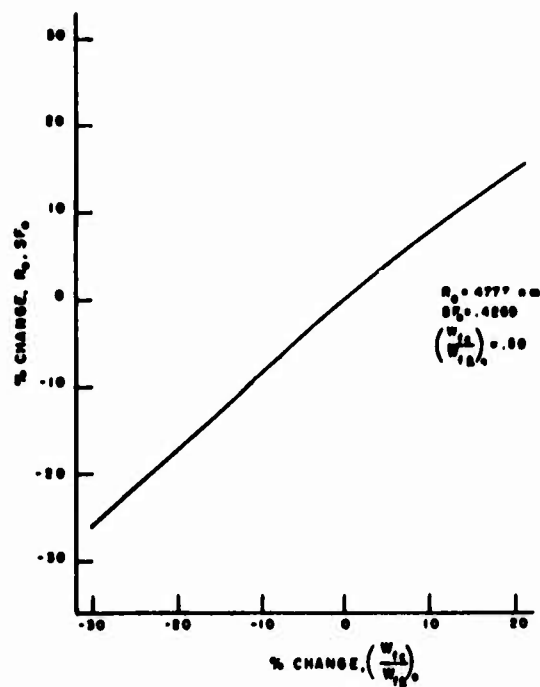


FIG 39 Fraction of Total Fuel Used in Cruise Influence on Performance

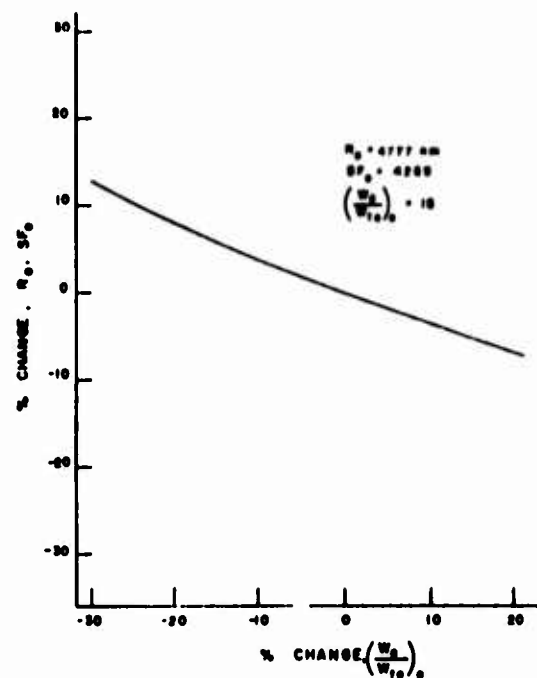


FIG 40 Equipment Weight Fraction Influence on Performance

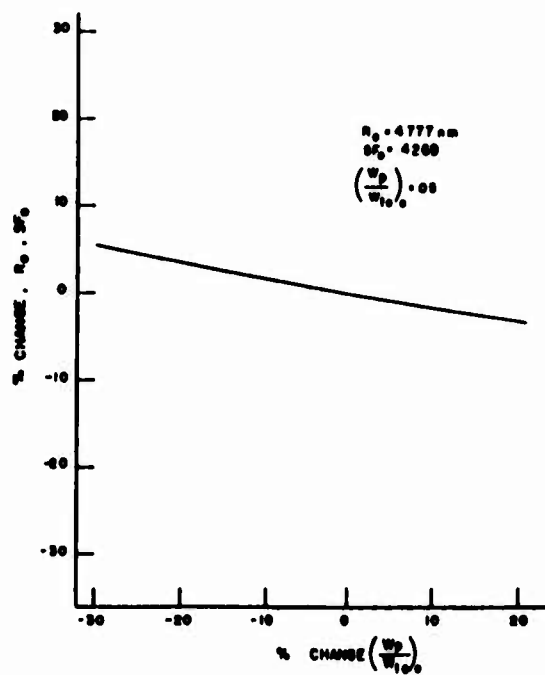


FIG 41 Payload Weight Fraction Influence on Performance

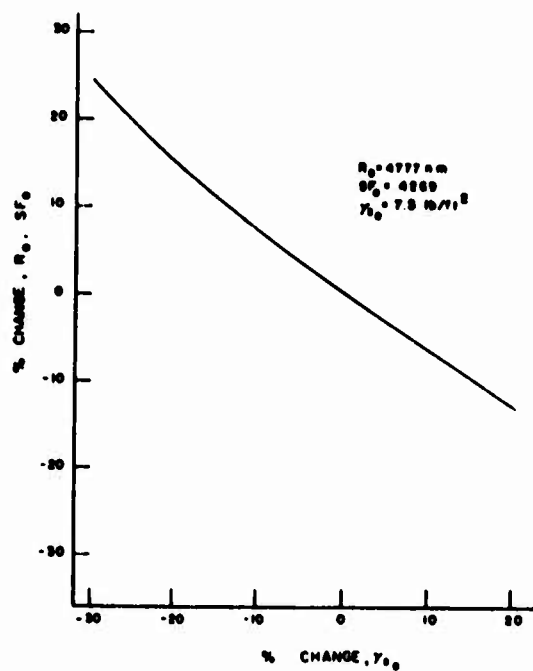


FIG 42 Wetted Area-Structural Weight Parameter Influence on Performance

Figure 39 illustrates that the cruise range increased as the amount of fuel consumed during cruise increased. This parameter, although producing a large change in the level of the range, does not have a large effect on the configuration of the optimized vehicle, since the level of range performance for all trial configurations are affected approximately the same.

Figure 41 illustrates that a 30 percent increase in the nominal payload weight fraction resulted in only about a 5 percent reduction in cruise range. Since the payload and fuel densities are equal, the scale factor (and hence the physical dimensions of the vehicle), remains constant with variations of the payload weight fraction. Passenger compartments and fuel tanks, for example, could be interchanged in a vehicle of fixed dimensions to accomplish a range-payload trade off.

Cruise range variation with the proportionality constant γ_s , which relates surface area of the vehicle to structural weight of the vehicle, is shown in Figure 42. In addition to having a pronounced effect on the level of the cruise range, the area-weight parameter also influences the optimized configuration. The greater the value of the area-weight parameter--the more important the volumetric efficiency, as reflected by the structural factor term of the Breguet range equation, becomes in determining the optimum configuration. Configurations optimized under the higher area-weight parameter would reflect greater volumetric efficiency at the expense of aerodynamic and or propulsive efficiency. For example, an area-weight constant greater than the nominal (7.5 lb/ft^2) would produce a sharper increase of structural factor with the first inlet ramp angle δ_1 shown in Figure 13. This would result in a higher value of the optimum inlet ramp angle. Thus, the qualitative influence of a value of the area-weight parameter other than the nominal on the configuration can be deduced from an analysis of the performance sensitivities of the independent configuration definition quantities. It can be noted from Figs 10 - 12 that the nominal value of area-weight parameter used in this study resulted in

structural weight fractions for the optimum configuration of $W_s / W_{to} = .258, .284, \text{ and } .296$. These values agree with the value of $W_s / W_{to} = .27$ used as a baseline in a wing structure study of a Mach 8 cruise vehicle (Ref 10).

Appendix D

Vehicle Cooling Requirements

Surfaces of the vehicle which cannot radiate to the surrounding space the heat produced by aerodynamic friction and combustion must be actively cooled. In addition, the aerodynamically sharp leading edges of the vehicle and wing (one-tenth of an inch diameter assumed for this investigation) may also require active cooling.

The purpose of this section is to develop the equations which indicate the amount of fuel needed to regeneratively cool internal surfaces of propulsive components, as well as the leading edges of the vehicle and wings. The fuel-air ratio required to cool the i -th surface is computed by equating the heat absorbing capacity available in the fuel diverted past the area A_{wi} of the i -th surface

$$\dot{Q}_{ci} = \dot{w}_{fi} \zeta = f_{ci} \dot{w}_{ai} \zeta = f_{ci} \rho_i V_i A_{pi} \zeta \quad (D1)$$

to the convective heat rate to the i -th surface

$$\dot{Q}_{hi} = \rho_i V_i S_{ti} (H_{abwi} - H_{wi}) A_{wi} \quad (D2)$$

If the cross sectioned area of the propulsive stream A_{pi} is evaluated adjacent to the i -th surface being cooled A_{wi} , then equating Eqs D1 and D2 results in an expression for the fuel-air ratio needed to provide coolant fuel for the i -th surface

$$f_{ci} = \left(\frac{A_w}{A_{pi}} \right) \frac{S_{ti}}{\zeta} (H_{abwi} - H_{wi}) \quad (D3)$$

or in terms of the equivalence ratio

$$\phi_{ci} = \frac{f_{ci}}{f_s} \quad (D4)$$

Equation D3 and D4 were solved for the following surfaces wetted by the propulsive stream: combustor, second inlet-forebody ramp and fences, inlet cowl and sides, nozzle cowl and sides,

nozzle-aftfuselage back to a point opposite the nozzle cowl.

The Stanton number was computed from the skin friction coefficient using Reynolds analogy $St_i = C_{f_i} / 2$.

The expression for laminar heat transfer to the stagnation line of an unswept cylindrical leading edge is given by (Ref 22)

$$\dot{q}_{stag} = 627.12 \left(\frac{V_1}{10^4} \right)^{3.15} \sqrt{\frac{\sigma}{r_n}} \quad (D5)$$

Assuming a cosine distribution of the heating rate about the stagnation point (Ref 23), the heat rate to the surface of the half cylinder leading edge is given by

$$\dot{Q}_{le} = 2 r_n b_{le} \dot{q}_{stag} \quad (D6)$$

where b_{le} is the combined leading edge width of the vehicle nose, cowl, and wings. The amount of regenerative cooling needed for the leading edges is found by equating Eq D1 and D6

$$\phi_{le} = \frac{2 r_n b_{le} \dot{q}_{stag}}{f_s \rho_1 V_1 \zeta A_c} \quad (D7)$$

Table IV illustrates for Mach 12 cruise, which is the most severe case, that cooling requirement for the leading edges of the vehicle was small compared to cooling requirements of the other components.

TABLE IV

Component Cooling Data For Mach 12 Configuration

| Component | Skin Friction Coefficient C_f | Reynolds Number R_e | Cooling Equivalence Ratio ϕ_c |
|------------------------------|---------------------------------------|-----------------------------|--|
| 2nd inlet ramp and fence | $.636 \times 10^{-3}$ | 1.45×10^8 | .0687 |
| inlet cowl and sides | 1.10×10^{-3} | 5.68×10^7 | .0778 |
| combustor and nozzle cowl | 1.6×10^{-3} | 3.19×10^7 | .1598 |
| nozzle-aftfuselage | 2.13×10^{-3} | 1.72×10^7 | .0238 |
| vehicle leading edges | ----- | ----- | .0015 |
| Total | | | .3316 |

Appendix E

Vehicle Design and Evaluation Computer Program

The problem of formulating the vehicle design and evaluation computer program can be broken down into three areas: first, the identification of quantities needed to define geometrically a configuration in the general class; second, the formulation of methods to predict the flow field properties about the configuration; and third, evaluation of the configuration performance.

Configuration Definition

Some of the quantities necessary to define geometrically a vehicle configuration of the class considered in this investigation can be arbitrarily chosen (independent) while others are calculated (dependent) from constraints. The purpose of this section is to formulate a set of independent configuration design variables and the constraint equations from which the dependent variables can be determined. In order to accomplish this aim, it is convenient to separate the vehicle into components: inlet-forebody, combustor, nozzle-aftfuselage, wing, and fuselage.

Inlet-forebody. In general, six geometric quantities ($l_1, l_2, l_3, h_4, \delta_1, \delta_2$) are required to define the inlet-forebody configuration as shown in Figure 43. It is essentially a two-dimensional double ramp inlet with a three-shock wave external compression system, with fences extending from the nose of the vehicle to the leading edge of the inlet cowl to contain the inlet air flow.

If it is assumed that at the design point the shock wave pattern is specified (shock on inlet cowl lip in this study), the number of variables needed to define the inlet configuration is reduced. Once the ramp angles (δ_1, δ_2) are specified, the flow turning angle of the inlet cowl ($\delta_3 = \delta_1 + \delta_2$) is known and the shock wave angles ($\theta_1, \theta_2, \theta_3$) can be calculated from the oblique shock relations (Ref 24). The variables

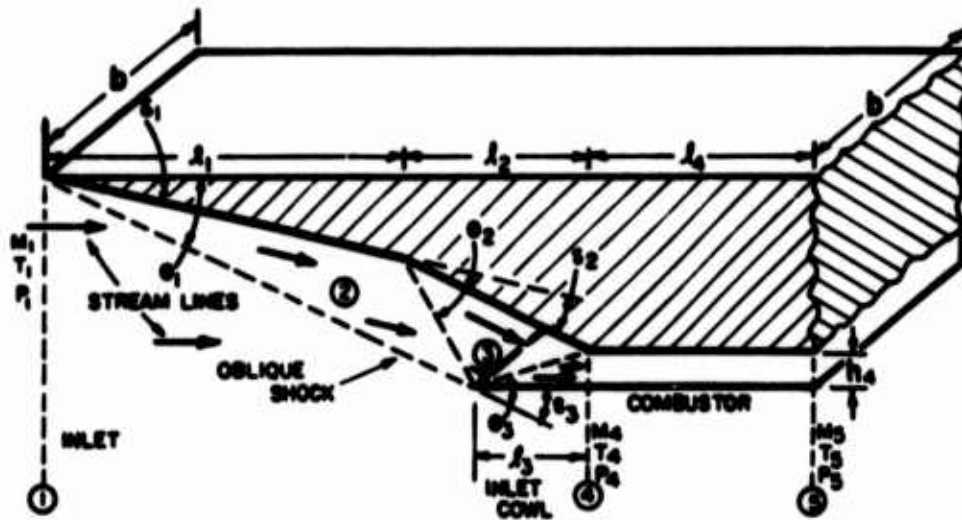


FIG 43 Three Shock Inlet and Combustor

(l_1, l_2, l_3, h_4) must then be related in such a manner that the shock waves from the double ramp inlet meet and reflect from the edge of the cowl to form a third shock wave. The third shock is cancelled by the appropriate turning angle at the top of the combustor entrance. The relationship among the variables (l_1, l_2, l_3, h_4) can be expressed in equation form by using the geometry of Figure 43:

$$(l_1 + l_2 - l_3) \tan \theta_1 = l_1 \tan \delta_1 + l_2 \tan (\delta_1 + \delta_2) + l_3 \tan (\theta_3 - \delta_3) \quad (E1)$$

$$(l_2 - l_3) \tan (\theta_2 + \delta_1) = l_2 \tan (\delta_1 + \delta_2) + l_3 \tan (\theta_3 - \delta_3) \quad (E2)$$

$$h_4 = l_3 \tan (\theta_3 - \delta_3) \quad (E3)$$

These are three constraint equations which relate four quantities; therefore, only one is independent. The quantity l_1 was arbitrarily chosen to be the independent variable.

Instead of specifying the ramp angle δ_2 , the temperature of the air entering the combustor T_4 was chosen as an independent quantity. This was done so that the temperature required for autoignition of

hydrogen and air (2000 R) could be placed directly as a lower bound on the design quantity T_4 to facilitate the optimization process. The ramp angle δ_2 necessary to produce the specified combustor entrance temperature T_4 was calculated using an iterative procedure along with the oblique shock equations. However, in order to calculate the shock angles θ and entrance combustor temperature T_4 , the flight Mach number M_1 , and ambient temperature T_1 must be specified in addition to the geometric variables, thus

$$\delta_2 = f(T_4, T_1, M_1, \delta_1) \quad (E4)$$

If the ambient temperature T_1 and ambient pressure p_1 are assumed to be functions of altitude H , such as in the set of relationships given by the standard atmosphere (Ref 25), then T_1 becomes a dependent quantity by reason of the following development.

Ambient pressure at start of cruise p_1 is determined from the constraint of equilibrium flight normal to the flight path; hence, p_1 becomes a function of lift L , weight at start of cruise W_i , and cruise velocity V_1 ,

$$p_1 = f(L, W_i, V_1) \quad (E5)$$

From the standard atmosphere relations

$$H = f(p_1) \quad (E6)$$

and

$$T_1 = f(H) \quad (E7)$$

therefore; the ambient temperature T_1 becomes a dependent quantity. Thus, the final set of independent configuration quantities needed to define the inlet was $(l_1, \delta_1, T_4, M_1)$.

A problem arises, however, in the sequences of calculation since it is necessary to know the ambient temperature T_1 before the ambient pressure p_1 is determined from the equilibrium flight constraint

(Equation E5), and T_1 from Equations E6 and E7. The problem can be solved by choosing a reference ambient temperature and assuming an isothermal atmosphere over the range of equilibrium flight altitudes, or by iteration on the equilibrium altitude. The standard atmosphere is isothermal only from approximately 36,000 ft to 88,000 ft at which point temperature begins to increase with altitude at approximately 1.6 R per thousand feet of altitude. Therefore, in the present study, an iteration was performed on the initial equilibrium altitude.

Combustor. The constant area combustor (Fig 43) is formed by the underside of the fuselage and a straight cowl located a distance h_4 from the fuselage. Both the underside of the fuselage and cowl are parallel with the body x-axis reference line. Independent configuration design quantities associated with the constant area combustor are the length l_4 and equivalence ratio ϕ . The length l_4 also determines the length of the vehicle midsection.

Equivalence ratio is defined in terms of stoichiometric fuel-air ratio f_s and actual fuel-air ratio f

$$\phi = \frac{f}{f_s} \quad (E8)$$

where the stoichiometric fuel-air ratio f_s is the ratio of fuel and air which results in all of the available fuel and air entering into the chemical reaction ($f_s = .0292$ for hydrogen). Thus, the equivalence ratio is a measure of the amount of energy added to the propulsive stream.

The equivalence ratio was designated an independent variable to facilitate the search for the optimum nozzle length and wing planform area. Normally, the fuel-air ratio is controlled (by a throttle) to produce unaccelerated flight for a fixed vehicle geometry. In the present formulation, however, it was more efficient, from a computational and search standpoint, to fix the amount of energy added to the propulsive stream and balance the resulting thrust with the drag from an appropriate amount of wing area. As explained in the algorithm at

the end of this section, wing area and performance parameters were calculated as each uprunning characteristic from the trailing edge of the cowl intersected the aftfuselage. Therefore, for one complete nozzle characteristic solution, which was expensive in terms of computation time, many nozzle lengths and wings area combinations were examined. In addition, the nozzle length was terminated when the range performance no longer increased with nozzle length when the situation occurred.

Nozzle-aftfuselage. The two-dimensional nozzle is formed by the aftfuselage and the rear portion of the cowl as shown in Figure 1. Three variables (l_5 , l_6 , δ_5) are needed to define the geometry of the nozzle-aftfuselage configuration. The maximum length of the nozzle l_5 was constrained such that the base height h_b was equal to or greater than zero. The minimum length of the nozzle cowl l_6 was also constrained as will be discussed later.

Wing. The variables (δ_9 , δ_{11} , b_w , c , δ_{12} , l_{ew} , AR , S_p) can be used to describe the wing geometry and location on the vehicle as shown in Figure 1. However, not all of the variables describing the wing are independent. The first two relationships between the wing variables comes from the definition of aspect ratio

$$AR = \frac{b_w^2}{S_p} \quad (E9)$$

and from the wing geometry

$$\begin{aligned} S_p &= c (c \tan \delta_{12}) + c (b_w - c \tan \delta_{12}) \\ &= c b_w \end{aligned} \quad (E10)$$

The wing planform area S_p was constrained such that the thrust and drag of the wing-body combination were equal (equilibrium flight along the flight path). Thus, the wing planform area S_p becomes a function of the thrust minus body drag

$$S_p = f(\text{thrust} - \text{body drag}) \quad (\text{E11})$$

The distance from the nose of the vehicle to the leading edge of the wing l_{ew} was determined by a trim requirement such that the pitching moment of the wing-body combination about the center of gravity of the fuselage be zero, hence

$$l_{ew} = f(\text{pitching moment about center of gravity}) \quad (\text{E12})$$

Attachment point of the wing leading edge and fuselage was constrained to be on the fuselage ($l_{ew} < l_b$) and was chosen to be in the xy-plane of the body axis system. The latter requirement restricted the angle of the top surface of the fuselage to positive values $\delta_7 \geq 0$.

Fuselage. Four independent configuration quantities (α , δ_7 , b , W_{t0}) are associated with the fuselage. The angle of attack α is the angle between the free stream flow direction and the vehicle reference axis shown in Figure 1. The angle δ_7 is the angle between the x-axis of the vehicle and the top surface of the fuselage; whereas, b is the width of the two-dimensional fuselage. Although δ_7 was constrained to positive values, a variable angle of attack allowed negative angles of incidence of the top surface of the fuselage with respect to the free stream flow.

The center of gravity of the two-dimensional fuselage, which can be controlled to some extent by placement of equipment and fuel in the vehicle, was assumed to coincide with the centroid of the profile area of the fuselage. It is shown in the results, however, that this assumption was not critical since the trim constraint did not seriously limit the range performance of the vehicle.

Although the geometrical shape of the vehicle is defined by the design quantities discussed thus far, the size or physical dimensions are arbitrary. The length of the first inlet ramp l_1 was used as the length scale factor for the vehicle. The equation for calculating l_1 (see Appendix C) can be written as a function of vehicle shape and

takeoff weight W_{to}

$$l_1 = f(\text{vehicle shape}, W_{to}) \quad (E13)$$

As in the case of altitude it is necessary to specify a reference length l_1 in order to calculate Reynolds numbers needed to determine skin friction in the force calculations.

Independent and dependent quantities used to define the geometrical model of the vehicle investigated in this study are summarized in Table IV. The number of independent quantities used to

| TABLE IV | |
|---|--------------------------|
| Configuration Definition Quantities | |
| <u>Independent Quantities</u> | |
| $\bar{X} = (\delta_1, T_4, \phi, l_4, \delta_5, l_5, l_6, \delta_9, \delta_{11}, \delta_{12}, AR, b, \delta_7, a, W_{to}, M_1)$ | |
| <u>Dependent Quantities</u> | |
| Variable | Calculated From Relation |
| l_2, l_3, h_4 | E1, E2, E3 |
| δ_2 | E4 |
| S_p | E11 |
| l_{ew} | E12 |
| b_w | E9 |
| c | E10 |
| P_1 | E5 |
| T_1 | E7 |
| l_1 | E13 |

define the model is analagous to the number of degrees of freedom used to describe a mechanical system with equality constraints. For the case depicted in Table IV there are $N = 27$ quantities defining the

configuration model and $m = 11$ equality constraint relations; thus, the degrees-of-freedom n of model is given by

$$n = N - m = 16 \quad (E14)$$

which corresponds to the number of independent configuration definition variables.

Flow Field Prediction

Once the vehicle configuration model is defined, the second major step in the problem formulation is the selection of methods to predict flow field properties about the vehicle. The distribution of flow field properties about the vehicle such as temperature, pressure, and velocity is needed to evaluate aerodynamic, propulsive, and cruise range performance. Inviscid flow field property distribution was determined using the following prediction techniques: adiabatic shock, Prandtl-Meyer expansion, one-dimensional constant - area supersonic heat addition, and method of characteristics. Local inviscid properties and turbulent (Ref 26) or laminar (Ref 27) skin friction laws, depending on the local Reynolds number, were used to compute the skin friction coefficient assuming a cold wall condition of $T_w = 2000R$. Appendix D contains typical values of skin friction coefficients obtained for various local surfaces.

Shock Expansion. Oblique shock relations were used to calculate flow field properties for compression surfaces and the Prandtl-Meyer relations for all expansion surfaces except the nozzle-aftfuselage. A constant value of 1.4 was used for the ratio of specific heats in both the oblique shock and Prandtl-Meyer relations.

The underside of the cowl, top surface of the fuselage, and the wing surfaces were treated as compression or expansion surfaces depending on the alignment of the particular surface with the flow. Due to the geometry of the above surfaces only one shock wave or expansion fan calculation per surface was required; however, the oblique shock

relations were also used to calculate the flow field properties in the inlet where three shock waves are involved. Initial conditions for the first shock wave calculation were the free stream properties, and a wedge angle equal to the sum of the first inlet ramp angle and angle of attack. Conditions behind the first shock were used as initial conditions for the second shock along with an assumed value of the second ramp angle δ_2 . Initial conditions for the third shock were the conditions behind the second shock, and since the cowl is parallel to the x-body axis, the wedge angle δ_3 for the third shock becomes the sum of the first and second ramp angles. An iterative procedure was used to determine the ramp angle δ_2 which produced the combustor entrance temperature T_4 specified as a design variable.

Constant Area Heat Addition. The combustor inviscid flow calculation procedure from station 4 to 5 (Fig 43) was based on the enthalpy method (Ref 28) and the one-dimensional, shockless, constant area heat addition relations for a mixture of gaseous hydrogen and air. Since the coolant fuel is also used for propulsion, initial conditions for the calculation assume that hydrogen fuel has a temperature of 2000 R (with no axial momentum component). Air flow properties at the combustor entrance were assumed equal to those at final inlet conditions. The gases leaving the combustor at station 5 were assumed to be in chemical equilibrium and produced by 100 per cent combustion efficiency.

Combustion products tables of Ref 29 along with the one-dimensional combustion equations were used to calculate combustor exit flow properties at station 5. This calculation requires an iteration on both the combustor exit pressure p_5 and temperature T_5 since dissociation and ionization of the combustion products are accounted for in the analysis.

Turbulent skin friction effects were superimposed on the inviscid pressure distribution to determine the combustor duct forces. Although

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Geometrically Eq E18 illustrates that the effect of combustor drag on the combustor pressure ratio increases with increasing combustor length to height ratio as expected. For typical values of combustor skin friction coefficients $C_{f_{av}} = .002$ and combustor length to height ratios of $l_4 / h_4 = 5$, the term $D_c / p_4 A_4 \gamma_4 M_4^2$ is small compared to unity. Thus, the effect of combustor drag on the pressure ratio is small for the configuration geometries considered.

Method of Characteristics. The method of characteristics (Ref 30) was used to determine the two-dimensional inviscid flow field properties in the nozzle. It was assumed that the flow entering the nozzle is uniform and that the total pressure remains constant during the expansion process. Mach number at the nozzle entrance was based on the frozen speed of sound determined from Ref 29 for the combustion products at combustor exit conditions. The frozen speed of sound at the combustor exit was also used to compute an effective specific heat ratio γ_5 which was then assumed to remain constant for the expansion process. The above method resulted in constant values of specific heats of $\gamma_5 \cong 1.27$ being used in the method of characteristic solutions.

A characteristic net was produced in the nozzle using a grid size of $\Delta\omega = 1.25^\circ$. As shown in Fig 44, expansion waves emanating from the sharp corner at 'a' can either strike the inner surface of the cowl and reflect to the aftfuselage, or miss the cowl and strike the free pressure boundary. Waves missing the cowl and striking the free pressure boundary do not reflect to the aftfuselage before the nozzle length is terminated; however, expansion waves emanating from the trailing edge of the cowl were accounted for when this condition existed. Due to programing considerations, the minimum length of the nozzle cowl was constrained to a length l_{6min} at which the first down running characteristic emanating from 'a' struck the nozzle cowl as shown in Figure 44.

the skin friction drag of the combustor duct was considered in the computation of vehicle drag, the effect of combustor skin friction drag on the inviscid flow field solution of the combustor was neglected, since the effect of skin friction drag on the inviscid solution can be shown to be small.

The pressure ratio p_5 / p_4 across the combustor can be written using the one-dimensional momentum equation and equations of state of a perfect gas at stations 4 and 5

$$\frac{p_5}{p_4} = \frac{\gamma_4 M_4^2 + 1 - \left(\frac{D_c}{p_4 A_4} \right)}{\gamma_5 M_5^2 + 1} \quad (\text{E15})$$

where D_c is the combustor friction drag. Factoring the numerator of Eq E15

$$\frac{p_5}{p_4} = \frac{\gamma_4 M_4^2 \left(1 + \frac{1}{\gamma_4 M_4^2} - \frac{D_c}{p_4 M_4^2 \gamma_4 A_4} \right)}{\gamma_5 M_5^2 + 1} \quad (\text{E16})$$

The term $D_c / p_4 A_4 \gamma_4 M_4^2$ can be compared to unity in order to determine under what conditions the pressure losses due to skin friction are small.

Since the ratio of combustor area wetted by the propulsive stream to the cross sectional area of the stream at station 4 is given by $((2h_4 l_4 + 2 l_4 b) / bh_4)$, the combustor drag term can be written as

$$\frac{D_c}{p_4 A_4 \gamma_4 M_4^2} = \left[\frac{\gamma_{c_{av}}}{\gamma_4} \frac{p_{c_{av}}}{p_4} \frac{M_{c_{av}}^2}{M_4^2} \right] \frac{l_4}{h_4} \left(1 + \frac{h_4}{b} \right) C_{f_{av}} \quad (\text{E17})$$

Since the term in the brackets on the right side of Eq E17 is on the order of unity, and the height of the combustor to the width of the vehicle is small, Eq E18 can be written

$$\frac{D_c}{p_4 A_4 \gamma_4 M_4^2} \approx \frac{l_4}{h_4} C_{f_{av}} \quad (\text{E18})$$

Geometrically Eq E18 illustrates that the effect of combustor drag on the combustor pressure ratio increases with increasing combustor length to height ratio as expected. For typical values of combustor skin friction coefficients $C_{f_{av}} = .002$ and combustor length to height ratios of $l_4 / h_4 = 5$, the term $D_c / p_4 A_4 \gamma_4 M_4^2$ is small compared to unity. Thus, the effect of combustor drag on the pressure ratio is small for the configuration geometries considered.

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Once the independent variable design set has been specified and the flow field properties calculated for a configuration, then the performance of the configuration can be evaluated. As pointed out in Chapter II, only the cruise segment of the mission profile was considered in the performance evaluation. This resulted in the Breguet range equation

developed in Appendix A becoming the criterion function in the determination of the optimum configuration. The purpose of this section is to develop the equations which relate the criterion function R to the independent configuration design variables. The Breguet range equation can be divided into two parts: (1) the so called Breguet range factor

$$RF = \frac{L}{D} \text{ Isp } \frac{V_1}{1 - V_1^2 / V_0^2} \quad (\text{E21})$$

and (2) the structural factor

$$SF = \ln (W_i / W_f) \quad (\text{E22})$$

Range Factor. The range factor is a measure of combined aerodynamic and propulsive performance for a given cruise speed. Equation (E21) is convenient for calculating range factor from L/D and Isp if these quantities are known for a configuration. However, to calculate the range factor from the aerodynamic flow field surrounding the vehicle and vehicle geometry, it is convenient to write Eq E21 in another form by retracing two steps in its development. Using the definition of specific impulse

$$\text{Isp} = \frac{T}{\dot{w}_f} \quad (\text{E23})$$

and the requirement for equilibrium flight along the flight path of the vehicle

$$T = D \quad (\text{E24})$$

Eq E21 becomes

$$RF = \frac{L}{\dot{w}_f} \frac{V_1}{1 - V_1^2 / V_0^2} \quad (\text{E25})$$

where the lift L is determined from the projection, in a direction normal to the flight path, of the pressure p and shear stress τ integrated over all wetted surfaces A_w of the configuration

$$L = \int \int_{\text{all surfaces}} (-p \hat{n} \cdot \hat{k}_w + \tau \hat{t} \cdot \hat{k}_w) dA \quad (\text{E26})$$

subject to the requirement for equilibrium flight along the flight path

$$\int \int_{\text{all surfaces}} (-p \hat{n} \cdot \hat{i}_w + \tau \hat{t} \cdot \hat{i}_w) dA = 0 \quad (\text{E27})$$

where \hat{i}_w , and \hat{k}_w are unit vectors along and normal to the flight path respectively, and \hat{n} and \hat{t} are the surface area normal and tangent unit vectors respectively.

In evaluating the range factor RF from Eq E25, bookkeeping decisions as to which forces are aerodynamic and which are propulsive are not important since the integration of the surface integrals in Eqs E26 and E27 is carried out over the entire surface of the configuration. However, if one wishes to determine classical aerodynamic and propulsive performance separately, a classification or division of the forces must be made. The definition of thrust used in this study is the projection of the nozzle force along the flight path given by

$$T = \int \int_{\text{nozzle surfaces}} (p \hat{n} \cdot \hat{i}_w - \tau \hat{t} \cdot \hat{i}_w) dA \quad (E28)$$

Once the I_{sp} has been determined then the L/D can be found from

$$L/D = (L/\dot{w}_f)I_{sp} \quad (E29)$$

Structural Factor. The purpose of the development in this section is to relate the structural factor $\ln(W_i/W_f)$ to configuration design variables in such a manner that the value of the structural factor is related to the configuration volumetric efficiency. Volumetric efficiency as defined in the report, is the ratio of volume V to surface area A_w .

If the takeoff weight W_{to} of the vehicle is assumed to be composed of payload W_p , equipment W_e , fuel W_{fl} and structure W_s , then the structural factor can be written as

$$\ln \frac{W_i}{W_f} = \ln \left[1 + \frac{\frac{W_{fc}}{W_{fl}} \frac{W_{fl}}{W_{to}}}{\frac{W_p}{W_{to}} + \frac{W_e}{W_{to}} + \frac{W_s}{W_{to}}} \right] \quad (E30)$$

where W_{fc} / W_{fl} is the fraction of total fuel used during cruise.

Two of the terms, W_{fl} / W_{to} and W_s / W_{to} , will be shown to be related to the volumetric efficiency of the vehicle. However, in order to evaluate the structural factor, from the set of independent design variables, it is necessary to introduce the structural set \bar{S} as discussed in Chapter II:

$$\bar{S} = \left[\frac{W_p}{W_{to}}, \frac{W_e}{W_{to}}, \frac{W_{fc}}{W_{to}}, \rho_p, \rho_e, \rho_f, \gamma_s \right] \quad (E31)$$

Neglecting the volume required for the structure, the enclosed volume of the vehicle is composed of volume required for payload V_p , equipment V_e , and fuel V_{fl} , thus

$$V = V_p + V_e + V_{fl} \quad (E32)$$

The takeoff weight is given by

$$W_{to} = W_p + W_e + W_{fl} + W_s \quad (E33)$$

Introducing ρ_p , ρ_e , and ρ_f from the structural parameter set \bar{S} , Eqs E32 and E33 become respectively

$$V_{fl} = V - \frac{W_p}{W_{to}} \frac{W_{to}}{\rho_p} - \frac{W_e}{W_{to}} \frac{W_{to}}{\rho_e} \quad (E34)$$

and

$$V_{fl} = \frac{1}{\rho_f} \left[W_{to} - \frac{W_p}{W_{to}} W_{to} - \frac{W_e}{W_{to}} W_{to} - W_s \right] \quad (E35)$$

The wetted area of the vehicle is introduced by assuming that the structural weight is a function of the wetted area--in particular a linear function

$$W_s = \gamma_s A_w \quad (E36)$$

Equating and rearranging Eqs E 34 and E35 along with Eq E36, allow the takeoff weight to be written as a function of vehicle volume V , wetted area A_w , and the structural parameter set \bar{S}

$$W_{to} = \frac{V + \gamma_s \left(\frac{A_w}{\rho_f} \right)}{\left(1 - \frac{W_p}{W_{to}} - \frac{W_e}{W_{to}} \right) \frac{1}{\rho_f} + \left(\frac{W_p}{W_{to}} \right) \frac{1}{\rho_p} + \left(\frac{W_e}{W_{to}} \right) \frac{1}{\rho_e}} \quad (E37)$$

For each configuration defined by a set \bar{X} of independent design variables, a relationship exists between V and A_w which, along with the additional relationship given by Eq E37, uniquely determines the volume and wetted area of the configuration. Once the vehicle volume V is known, the total fuel volume V_{fl} can be determined from Eq E34 which along with the fuel density ρ_f enables the total fuel fraction W_{fl} / W_{to} to be calculated.

The structural fraction W_s / W_{to} can be determined from Eq E36 knowing the wetted area A_w and proportionality constant γ_s . All of the terms in the structural factor are now known. The structural factor, and hence, the cruise range R can be evaluated given the set \bar{X} of independent design quantities and the set \bar{S} of structural parameters.

Algorithm. This appendix is summarized by the computational algorithm of the design and evaluation computer program. The purpose of the algorithm is to relate the general formulation contained in this appendix to the subroutines contained in the listing of the program which details the equations used in the vehicle design and evaluation computer program.

For a given set of independent design variables the configuration and performance of a vehicle is determined by the following procedure:

1. Input data including the independent design variables is contained in MAIN.

2. Initial conditions for flow field calculations are set in subroutine RANGE.
3. Inlet-forebody flow field parameters and shock wave angles are calculated in subroutine SHOCK using oblique shock relations.
4. Inlet configuration and combustor height are computed in CONFIG using the constraint of shock on lip.
5. Flow field properties are determined for the underside of cowl, fuselage top, and wing surfaces, by SHOCK or PRANT depending on whether the component is a compression or expansion surface respectively.
6. One-dimensional supersonic heat addition equations are solved in CBMST to determine the inviscid flow properties at the combustor exit.
7. Inviscid and viscid (subroutine SKINF) forces and moment are calculated by subroutine FAM for the inlet, cowl, fuselage top and sides back to the point where the nozzle-aftfuselage begins.
8. Inviscid nozzle flow properties are calculated by the method of characteristics in subroutines NOZZL, STARTC, PMSBR, and LPS.
9. Force and moments are integrated on the nozzle-aftfuselage by subroutine FAM. Integration step size is the axial distance between the intersection of the nozzle-aftfuselage and adjacent up-running characteristics.
10. The following computations are made in subroutine FAM whenever an up-running characteristic strikes the nozzle-aftfuselage.
 - (a) Thrust forces produced by the nozzle and the drag of the inlet, cowl, and fuselage are compared.
 - (b) If the thrust is less than body drag, another up-running characteristic is computed which increases the nozzle length.
 - (c) Steps (a) and (b) continue until: (i) the nozzle reaches the length specified in the design set, or (ii) the nozzle surface intersects the top surface of fuselage (zero base) or (iii) the thrust is greater than the body drag. If conditions (i) or (ii) occur, the

calculation is terminated and the cruise range solution does not exist for the selected set of configuration variables.

(d) If the thrust is greater than body drag, wing forces and moments are determined. A wing area is determined such that the drag of the wing-body combination equals the thrust produced by the nozzle.

(e) The center of force (for the wing-body combination) and center of gravity (based on the center of area of the body profile) is determined. Placement of the wing on the body is then calculated such that the sum of the moments about the center of gravity is zero (trimmed flight condition). If the trim requires a placement of the wing such that the wing is off the body, the calculation returns to step 10 (c).

(f) Scaling or sizing the vehicle is made by relating vehicle volume and wetted area to takeoff weight and the members of the structural parameter set \bar{S} .

(g) Ambient pressure at initial cruise altitude is computed from the requirement for equilibrium flight normal to the flight path.

(h) Structural, aerodynamic, propulsive, and cruise range performance (criterion function) are calculated.

(i) Another up-running characteristic is computed which increases the nozzle length.

(j) Steps (d) through (i) are repeated until one of the following case termination conditions are reached: (I) cruise range decreases with increasing nozzle length, (II) nozzle length greater than length specified in design set or (III) intersection of the nozzle-aftfuselage surface and fuselage upper surface (zero base).

11. If the case is terminated by condition (I), the performance value of the point before the decrease in performance was noted is used as the final performance value.

12. If the case is terminated by conditions (II) or (III), between the current point (exceeded constraint) and previous point (within)

constraint), a linear interpolation is performed in subroutine NOZZL to adjust configuration performance and parameters to satisfy the constraints.

13. Cooling requirements of the combustor and portions of the inlet and nozzle as well as the leading edge are determined in subroutine COOL.

14. Configuration performance and parameters are printed in subroutine MAIN.

```

DELO(11)=7.41
DELC(12)=0.
C REFERENCE QUANTITIES TO COMPUTE SKIN FRICTION AND
PASPO=.2403
C FREE STREAM PROPERTIES--MF=ALITUDE(FT), DI=SCALE FACTOR(FT)
MF=1.15E3
DI=0.02
C DIAPETER OF LEADING EDGE(FT)
DI=0.1712
C CAV95--COEFFICIENT FOR DETERMINING AVERAGE COMBUSTOR PROPERTIES
CAV95=.5
C COMPUTATION CONTROL CAMES
PCOM=0
C PCOM=HING PLACEMENT CONTROL--IF MCOM NOT EQUAL 1--COMPLETE HING
C PLACEMENT PREP MOMENT ABOUT CG EQUAL ZERO REQUIREMENT-MCOM EQUAL 1
C PIR=HING POSITION
C ST(2)=SKIN FRICTION COMPUTATION CONTROL--IF ST(2) LESS THAN ZERO
C COMPUTE SKIN FRICTION FOR LOCAL SURFACES--ST(2) EQUAL ZERO OR GREATER--
C USE VALUE OF ST(2) AS SKIN FRICTION COEFFICIENT FOR ALL SURFACES
ST(2)=0.2
MCOM=2
ALPHAR=0.01
DELS=0.01
PIZ=0.5
PALT=0.
M=0.
L=0.
MCOM=0
C DETERMINE CONFIGURATION AND PERFORMANCE FOR ABOVE INPUT DATA
CALL RANGE
C DETERMINE REGENERATIVE COOLING REQUIREMENTS
CALL CCOL(INEC)
STOP
END

SUBROUTINE RANGE
SURFUTIME RANGE
C THIS SUBROUTINE CALLS COMPUTATIONAL SUBROUTINES IN PROPER SEQUENCE
C WITH PROPER INITIAL CONDITIONS
COMMON /VAR/ M(13), DELC(13), LAM(13), TETA(13), MCM, M2(13),
X(10), P, Y4, Y5, CFX(13), CFT(13), CEL5, ALPHAR, CACAP, TLAP(13), PCESS,
ZER, APS, P
REAL P, M2, LAM
DATA PI, P2 / 0.17453292, 57.295779 /
RCE(12)=2.0F1
DEG(12)=2.0F2
C X(13), LAM(13), TLAP(13) ARE NON-DIMENSIONAL LONGITUDINAL LENGTH--L(13)/DI,
C PRESSURE--P(13)/PI, AND TEMPERATURE--T(13)/T1 RESPECTIVELY
C COMPUTE INLET FLOW FIELD PROPERTIES
C INITIAL CONDITIONS
103 X(13)=1.
LAP(13)=1.
TLAP(13)=1.
C ADJUST FIRST RAMP ANGLE FOR ANGLE OF ATTACK
ALPHAR=DEG(13)
DELC(13)=RADI(13)*RADI(13)
DELC(12)=RADI(13)
C SOLVE CLIQUE SHOCK WAVE EQUATIONS FOR INLET SHOCK SYSTEM
CALL SHOCK(13, 0.17)
DELC(12)=DEG(13)

```

APPENDIX E (CONTINUED)

VEHICLE DESIGN AND EVALUATION PROGRAM COMPUTER LISTING

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SUBROUTINE RANGE
SURFUTIME RANGE
C THIS SUBROUTINE CALLS COMPUTATIONAL SUBROUTINES IN PROPER SEQUENCE
C WITH PROPER INITIAL CONDITIONS
COMMON /VAR/ M(13), DELC(13), LAM(13), TETA(13), MCM, M2(13),
X(10), P, Y4, Y5, CFX(13), CFT(13), CEL5, ALPHAR, CACAP, TLAP(13), PCESS,
ZER, APS, P
REAL P, M2, LAM
DATA PI, P2 / 0.17453292, 57.295779 /
RCE(12)=2.0F1
DEG(12)=2.0F2
C X(13), LAM(13), TLAP(13) ARE NON-DIMENSIONAL LONGITUDINAL LENGTH--L(13)/DI,
C PRESSURE--P(13)/PI, AND TEMPERATURE--T(13)/T1 RESPECTIVELY
C COMPUTE INLET FLOW FIELD PROPERTIES
C INITIAL CONDITIONS
103 X(13)=1.
LAP(13)=1.
TLAP(13)=1.
C ADJUST FIRST RAMP ANGLE FOR ANGLE OF ATTACK
ALPHAR=DEG(13)
DELC(13)=RADI(13)*RADI(13)
DELC(12)=RADI(13)
C SOLVE CLIQUE SHOCK WAVE EQUATIONS FOR INLET SHOCK SYSTEM
CALL SHOCK(13, 0.17)
DELC(12)=DEG(13)

```



```

8FC112, P0112, C3, P0TP, COLEBC, CLN, VMD12B, ANGDI2, XCPLEV, CDB, CLB, TML,
$WDC, CUP(133), CKR
DIMENSION PMB(100, 4, 35), P0(4), P1(4), P2(4), P3(4), LPT(133, 2)
EQUIVALENCE (LPT(1,1), SEFCY)
REAL P, M2, LAM, LPT
PRTOFIA, B1 = 1 - ((B-1.) / (B+1.)) * A**2 / ((B-1.))
EMACH, FIA, B1 = SCRT(((12. * (A**2)) / (B+1.)) / (1. - ((B-1.) / (B+1.)) * (A**2)))
EMASTFIA, B1 = SCRT((0.5 * (B+1.)) * (A**2)) / (1. + 0.5 * (B-1.)) * (A**2)))
N6=0
I=0
K=0
DXCWL=XCWL
P15=.5
BETA=0.
X5L=0.
Y5L=0.
PAL'L=0.
DCL=0.
GAMMAP=GAMS
AL=0.
EP10=AP5
EPSIP=EMASTFIEMLP, GAMMAP
600 ANGLE=-CELC(5)
N1=1
C COMPUTE CCAN RUNNING CHARACTERISTICS EMANATING FROM NOZZLE EXPANSION
C CCANER
CALL PWSE (1, GAMMAP, EMIP, N1, ANGLE, BETA, K, PMB)
C COMPUTE STARTING LINE FOR CHARACTERISTICS SOLUTION
CALL STARTC (EMIP, EMSP, PMB)
C COMPUTE PROPERTIES AT LATTICE POINTS ALONG FIRST 9 UPRUNNING
C CHARACTERISTICS ORIGINATING ON STARTING LINE
DC 5 I=9
DC 6 J=1, K
DC 7 L=1, 4
P1(1)=PMB(J, L, I)
P2(1)=PMB(J+1, L, I-1)
7 C COMPUTE FLOW PROPERTIES AT INTERIOR LATTICE POINTS OF FLOW FIELD
CALL LPS (1, GAMMAP, ANGLE, P1, P2, P3)
DC 8 L=1, 4
DC 9 L=1, 4
P2(1)=PMB(J+1, L, I-1)
P2(1)=PMB(J+1, L, I-1)
9 C COMPUTE FLOW PROPERTIES AT LATTICE POINT ON NOZZLE-FUSELAGE SURFACE
CALL LPS (2, GAMMAP, ANGLE, P1, P2, P3)
L1=K+2
L5=1
PP4=57.2957795 * P3(4)
PP3=EMACHFIP(133), GAMMAP
11 DC 12 L=1, 4
12 PPRK+2, L, I=P3(L)
5 K=K+1
N1=1
C COMPUTE PROPERTIES AT LATTICE POINTS ALONG UPRUNNING CHARACTERISTICS
C ORIGINATING FROM NOZZLE COWL OR TRAILING EDGE OF NOZZLE CCWL
DC 10 J1=N1, 10000
DC 70 L=1, 4
P1(1)=PMB(J1, L, I)
P2(1)=PMB(J1+1, L, I)
70 C COMPUTE PROPERTIES AT SURFACE OF NOZZLE COWL
CALL LPS (3, GAMMAP, ANGLE, P1, P2, P3)

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F2=ICJ2/(V4*V4*Q11) * (1 + 4 * V4 * V4 / GJ2 + MFAB * ER * FST) - H5 * GJ2 / (V4 * V4)
C TEST TO DETERMINE IF VELOCITY RATIO WILL SATISFY MOMENTUM, ENERGY AND
C CONTINUITY EQUATIONS SIMULTANEOUSLY
F1=V4*V4*Q11
F1=1-F2
TEARS(F, LT, 1, 5=3) GO TO 8
C COMPUTE NEW TAIL TEMPERATURE USING NEWTON-RAPHSON TECHNIQUE
NS=NCI/PAIAC
FCTA=(1-R5*P3)/(1+SCRT(PAR-4. * A * C1))-R5/A+GJ2*CP5/(V4*V4)
C1=1.
TEMP5=TEMP05-C1 * F / FCTOT
C CONTINUE TEMPERATURE ITERATION
NA=NA+1
IF (NA .GT. 40) GO TO 40
CC TC 0
V5=V5*V4*V4
P5P4=L0(ITEMPS/TEMP4) * (BARAK4/BARMS1) * (V4/V5)
P5P4
P5C=P4 * P
IF (P5C / (P5C - P5)) / P51 .LT. .001 GO TO 11
C CONTINUE PRESSURE ITERATION
P5=P5 * (P5C - P5) * .9
P5=P5+1
IF (P5 .GT. 40) GO TO 40
CC TC 12
CONTINUE
11 C DETERMINE FROZEN SPEED OF SOUND
CALL SPEEDIT * P5, ER, P5, CAF5)
C SUBROUTINE SPEED CONTAINS FROZEN SPEED OF SOUND OF COMBUSTION GASES
C AS A FUNCTION OF EQUIVALENCE RATIO, PRESSURE, AND TEMPERATURE (REF 29)
C COMPUTE MACH NUMBER STATION 5
AP5=V5/CAFS
C COMPUTE RATIO OF SPECIFIC HEATS AT STATION 5
GAMS=CAFS * CAF5 / (R5 * TEMP5)
PC5=P5 * PARTIO (AP5, GAMS)
MINF5=PS
IF (GMS .LT. 1.05) GO TO 50
RETURN
40 WRITE (1, 31)
51 FORBATT(2X, 4, JHPRESS OR TEMP FAILED TO CONVERGE AT ST 5, /)
RETURN 1
50 WRITE (1, 51)
51 FORBATT(2X, 16, COMBUSTEP CHOKED, /)
RETURN 1
END
11PTC ACZZ
SUBROUTINE NOZZLE(1)
C THIS SUBROUTINE CONTROLS NOZZLE FLOW FIELD AND FORCE CALCULATIONS
COMMON /V4K/ (133), DEL(133), DELD(133), LAM(133), THETA(133), XCWL, P2(133),
ZEP, AP, P, V4, V5, CFA(133), CFY(133), DEL5, ALPHA, CACAP, TILAM(133), FPCSS,
COMMON /V4A2/ (12), PASPR, MF, PDES, PALT, NTJD, NTOT, CWT, PALT, CPALT, X5D
COMMON /ADCI/ (1), CL, L, WTO
COMMON /JAMEC/ TEMPS, V4, V5, MINF5, GAMS, BDI, CAV45, RS, AP6, TEMP65
COMMON /VEE/ CC
COMMON /PPST/ SEFCY, STFRAG, XCM, YCM, BWRG, ANNG, VTD12B, ATGD12, COSF,
$XMLEV, PCH, KW, AISPT, AISPL, ALDCL, ALUOT, RANF, FDN, CDB, COLEN,

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C TEST TC DETERMINE IF X-COORDINATE OF PROPOSED UPBURNING
C CHARACTERISTIC IS GREATER THAN DESIRED LENGTH OF THE NOZZLE COML
IF (P3(1).GE.DXCML) GO TO 31
C DETERMINE FORCE AND MOMENTS OF NOZZLE COML
CALL FAN (1,GAMMAP,P1,P2,P3,I,N6,$777,$778)
L2=J1+1
I6=I+1
P4=57.2957795*P3(4)
P3=EPACMF(P3(3),GAMMAP)
DC 80 L=1.4
N2=AL+1
DC 60 J2=N2,K
DC 71 L=1.4
P1(L)=PMB(J2,L,I+1)
P1(L)=PMB(J2,L,I+1)
C INTERIOR LATTICE POINT
CALL LPS (1,GAMMAP,ANGLE,P1,P2,P3)
L2=J2+1
I7=I+1
P3=EPACMF(P3(3),GAMMAP)
P4=57.2957795*P3(4)
DC 81 L=1.4
P1(L)=PMB(J2+1,L,I+1)
P1(L)=PMB(J2+1,L,I+1)
C SAVE CLD VALUES OF CONFIGURATION AND PERFORMANCE PARAMETERS
PALT=PALT
RL=R
XSL=X(5)
DCL=CC
VSL=VS
DC 93 L=1.33
LPT(L,2)=LPT(L,1)
C DETERMINE FORCES AND MOMENTS ACTING ON THE VEHICLE
CALL FAN (2,GAMMAP,P1,P2,P3,I,N6,$777,$778)
K5=K+2
I7=I+1
P4=57.2957795*P3(4)
P3=EPACMF(P3(3),GAMMAP)
DC 121 L=1.4
P1(L)=PMB(J2,L,I+1)
P1(L)=PMB(J2,L,I+1)
C TEST TC DETERMINE IF CONSTRAINTS HAVE BEEN EXCEEDED
IF (V5.LE.0.001) GO TO 500
IF (X(5).GT.X50) GO TO 502
GC TC IS
C LINEAR INTERPOLATION BETWEEN CURRENT AND PREVIOUS POINT
C 400 RETURN
C 500 ZERC BASE SOLUTION
DVS=VSL/(V5-V5L)
DCL=CC*(DC-DCL)+CVS
PALT=PALT*(PALT-PALT)+CVS
X5=XSL*(X(5)-X5L)+CVS
IF (KCK.EQ.1) GO TO 699
RS=RL*(R-RL)+CVS
DC 501 L=1.33
LPT(L,1)=LPT(L,2)+(LPT(L,1)-LPT(L,2))*DVS
P=RS
V5=D.
X(5)=X55
698 WRITE (6,700) PALT,PDES,DCS,C1
IF (X(5).GT.X5C) GO TO 502
700 FCWPAT (2,X,18)*ZERO BASE SOLUTION//2X,19*PALT,PALTO,DIC,D1
81P4E15.7//)
RETURN
C NOZZLE LENGTH EQUAL DESIRED LENGTH
699 IF (ARS(X(5)-X5C) .LE. 0.05) GO TO 698

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C TEST TC DETERMINE IF X-COORDINATE OF PROPOSED UPBURNING
C CHARACTERISTIC IS GREATER THAN DESIRED LENGTH OF THE NOZZLE COML
IF (P3(1).GE.DXCML) GO TO 31
C DETERMINE FORCE AND MOMENTS OF NOZZLE COML
CALL FAN (1,GAMMAP,P1,P2,P3,I,N6,$777,$778)
L2=J1+1
I6=I+1
P4=57.2957795*P3(4)
P3=EPACMF(P3(3),GAMMAP)
DC 80 L=1.4
N2=AL+1
DC 60 J2=N2,K
DC 71 L=1.4
P1(L)=PMB(J2,L,I+1)
P1(L)=PMB(J2,L,I+1)
C INTERIOR LATTICE POINT
CALL LPS (1,GAMMAP,ANGLE,P1,P2,P3)
L2=J2+1
I7=I+1
P3=EPACMF(P3(3),GAMMAP)
P4=57.2957795*P3(4)
DC 81 L=1.4
P1(L)=PMB(J2+1,L,I+1)
P1(L)=PMB(J2+1,L,I+1)
C SAVE CLD VALUES OF CONFIGURATION AND PERFORMANCE PARAMETERS
PALT=PALT
RL=R
XSL=X(5)
DCL=CC
VSL=VS
DC 90 L=1.4
P2(L)=PMB(K+1,L,I)
P2(L)=PMB(K+1,L,I+1)
C NOZZLE-POUSELAGE SURFACE LATTICE PCINT
CALL LPS (2,GAMMAP,ANGLE,P1,P2,P3)
C SA E CLD VALUES OF CONFIGURATION AND PERFORMANCE PARAMETERS
DCL=DC
XSL=X(5)
VSL=VS
PALT=PALT
DC 92 L=1.33
LPT(L,2)=LPT(L,1)
C DETERMINE FORCES AND MOMENTS ACTING ON THE VEHICLE
CALL FAN (2,GAMMAP,P1,P2,P3,I,N6,$777,$778)
L1=K+2
I5=I+1
P4=57.2957795*P3(4)
P3=EPACMF(P3(3),GAMMAP)
DC 120 L=1.4
P1(L)=PMB(J2,L,I+1)
P1(L)=PMB(J2,L,I+1)
C TEST TC DETERMINE IF GEOMETRIC CONSTRAINTS(ZERO BASE OR NOZZLE
C LENGTH EXCEEDING DESIRED LENGTH) HAVE BEEN EXCEEDED
IF (V5.LE.0.001) GO TO 500
IF (X(5).GT.X50) GO TO 502
I=I+1
N1=N1+1
50 K=K+1
GC TC 400
31 CCONTINUE
K=K
C COMPUTE FORCE AND MOMENT ON NOZZLE COML
CALL FAN (3,GAMMAP,P1,P2,P3,I,N6,$777,$778)
C COMPUTE UPBURNING CHARACTERISTICS EMANATING FROM CENTERED EXPANSION
C ABOUT TRAILING EDGE OF THE NOZZLE COML
CALL FMSBR (2,GAMMAP,ENIP,N1,ANGLE,BETA,K,PMB)

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100 FORMAT(9F13.6)
1 CONTINUE
RETURN
END

SUBROUTINE LPSIKO,GAMMAP,ANGLE,P1,P2,P3)
C THIS SUBROUTINE COMPUTES PROPERTIES AT LATTICE POINTS IN NOZZLE FLOW FIELD
C INCLINATION FOLLOWS THAT OF SHAPING REF 30)

COMMON /VAR/M(13),DEL(13),DEL(13),LAW(13),THETA(13),XWLM,M2(13),
18(13),P,YA,Y5,CPR(13),CFV(13),CELS,ALPHAR,CACAP,TLAM(13),FPCSS,
ZER,AP5,P
DIMENSION P1(4),P2(4),P3(4)
REAL P,M2,LAM
EPCAPF(A,B)=SQRT((12.0*(A+2))/((B+1.1)/(1.0-1.1)/(B+1.1))+(A+2)))
ALPAPF(C)=ATAN(SQRT(1.1/(C+2-1.1)))
OPECAPF(A,B)=SQRT((B+1.1)/(B-1.1)))*ATAN(SQRT((A+2-1.1)/(B+1.1)/(B-1.1)
1-A+2)))-ATAN(SQRT((B+1.1)/(B-1.1))+(A+2-1.1)/(B+1.1)/(B-1.1)-A+2)
2)))
D=57.29577
C PHYSICAL COORDINATES OF POINTS (1) AND (2)
R1=0.111
R2=0.211
V1=0.112
V2=0.212
C M STAR AT POINTS (1) AND (2)
EPS1=0.1(13)
EPS2=0.2(13)
C INCLINATION OF FLOW DIRECTION AT POINTS (1) AND (2)
THET1=0.1(4)
THET2=0.2(4)
C CHARACTERISTIC COORDINATES AT POINTS (1) AND (2)
DPEG1=CMEGAF(EMS1,GAMMAP)
DPEG2=CMEGAF(EMS2,GAMMAP)
E11=11200.0*CEG10C*THET10/2.
E112=11000.0*PEG20C*THET20/2.
GC TC 11,2,31,44
C MODOGRAPH SOLUTION OF INTERIOR POINT (3) KNOWING POSITION AND PROPERTIES
1 AT POINTS (1) AND (2)
1 E13=E11
E113=E112
THET3=(E13+E113-1000.1/D
CPEG3=(E13-E113)/D
C INITIAL TRIAL VALUE FOR M STAR AT (3)
EPS3=(EMS1+EMS2)/2.
CPEG=CMEGAF(EMS3,GAMMAP)
DCEG=CMEG3-CMEG
T1=ABS(CMEG)
IF(T1-1.E-4)22,22,32
C=(GAPPAP+1.1)/(GAMMAP-1.1)
T3=CEG/12.0*SQRT((T2-1.1)*SQRT(C-1))
T4=1.-1.1/(T2*SQRT(C))
F1=T3*0.74
DMS32=CMEG/F1
EPS32=T2+2.0*CMS32
EPS3=SCRT(EMS32)
GC TC 42
22 ALPH3=ALPHAF(EMACHF(EMS3,GAMMAP),GAMMAP)
ALPH1=ALPHAF(EMACHF(EMS1,GAMMAP),GAMMAP)
ALPH2=ALPHAF(EMACHF(EMS2,GAMMAP),GAMMAP)
ANG1=(THET3-ALPH3+THET1-ALPH1)/2.
ANG2=(THET3-ALPH3+THET2-ALPH2)/2.
BI-TAN(ANG1)
BI-TAN(ANG2)
GC TC 39
20 ALPH3=ALPHAF(EMACHF(EMS3,GAMMAP),GAMMAP)
ALPH1=ALPHAF(EMACHF(EMS1,GAMMAP),GAMMAP)
ALPH2=ALPHAF(EMACHF(EMS2,GAMMAP),GAMMAP)
ANG1=(THET3-ALPH3+THET1-ALPH1)/2.
ANG2=(THET3-ALPH3+THET2-ALPH2)/2.
C AVERAGE SLOPES OF MACH LINES FROM (1-3) AND (2-3)
BI-TAN(ANG1)
BI-TAN(ANG2)
GC TC 39
C MODOGRAPH SOLUTION OF NOZZLE-FUSELAGE BOUNDARY POINT WHERE FLOW ANGLE
AND ONE COORDINATE IN MODOGRAPH PLANE IS KNOWN
2 THET3=ANGLE/D
E13=E11
DPEG3=(12.0*E13-1000.-THET3*01)/C
EPS3=(EMS1+EMS2)/2.
CPEG=CMEGAF(EMS3,GAMMAP)
DCEG=CMEG3-CMEG
T1=ABS(CMEG)
IF(T1-1.E-4)21,21,31
C=(GAPPAP+1.1)/(GAMMAP-1.1)
T3=CEG/12.0*SQRT((T2-1.1)*SQRT(C-1))
T4=1.-1.1/(T2*SQRT(C))
F1=T3*0.74
DMS32=CMEG/F1
EPS32=T2+2.0*CMS32
EPS3=SCRT(EMS32)
GC TO 41
21 ALPH3=ALPHAF(EMACHF(EMS3,GAMMAP),GAMMAP)
ALPH1=ALPHAF(EMACHF(EMS1,GAMMAP),GAMMAP)
ALPH2=ALPHAF(EMACHF(EMS2,GAMMAP),GAMMAP)
ANG1=(THET3-ALPH3+THET1-ALPH1)/2.
ANG2=(THET3-ALPH3+THET2-ALPH2)/2.
BI-TAN(ANG1)
BI-TAN(ANG2)
GC TC 39
C MODOGRAPH SOLUTION OF NOZZLE COHL BOUNDARY POINT WHERE FLOW ANGLE
AND ONE COORDINATE IN MODOGRAPH PLANE IS KNOWN
3 THET3=0.
E13=E112
CPEG3=(11000.-2*0.113+THET3)/D
EPS3=(EMS1+EMS2)/2.
CPEG=CMEGAF(EMS3,GAMMAP)
DCEG=CMEG3-CMEG
T1=ABS(CMEG)
IF(T1-1.E-4)22,22,32
C=(GAPPAP+1.1)/(GAMMAP-1.1)
T3=CEG/12.0*SQRT((T2-1.1)*SQRT(C-1))
T4=1.-1.1/(T2*SQRT(C))
F1=T3*0.74
DMS32=CMEG/F1
EPS32=T2+2.0*CMS32
EPS3=SCRT(EMS32)
GC TC 42
22 ALPH3=ALPHAF(EMACHF(EMS3,GAMMAP),GAMMAP)
ALPH1=ALPHAF(EMACHF(EMS1,GAMMAP),GAMMAP)
ALPH2=ALPHAF(EMACHF(EMS2,GAMMAP),GAMMAP)
ANG1=(THET3-ALPH3+THET1-ALPH1)/2.
ANG2=(THET3-ALPH3+THET2-ALPH2)/2.
BI-TAN(ANG1)
BI-TAN(ANG2)
GC TC 42

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CXL2=CFXL2*(CELY/2.*PI2I2I)
CPUP=CF
GC TC 65
C CENTRAUE
CPYL2=PRPAV*CEL2-TAN(P3I4I)*PRPAV*DEL*GM*EPACH3*0.2.*CF/2.
CPYL2=CFYL2*(CELY/2.*PI2I2I)
CPYL2=PRPAV*CEL2-TAN(P3I4I)*PRPAV*DEL*GM*EPACH3*0.2.*CF/2.
CPYL2=CFXL2*(CELY/2.*PI2I2I)
CPYL2=0.
C COMPUTE NOZZLE FENCE DRAG
DPAV=(EMACH3*0.2.*PRPAV*CPSTI/2.
AUNCZ=IV4*X(1I)*TAN(CELSI)*X(5I)/BDI
CPNCZ=CF
FACZ=CPAV*CFNOZ*GM*(AMNOZ-AMNOZL)/12.*V4)
CPYL2=CFYL2*FNOZ
AUNCZL=AMNCZ
CPYL2=0.
CPYL2=0.
GC TC 65
C FORCE AND MOMENT CALCULATION WHEN FIRST DOWN RUNNING CHARACTERISTIC
C STRIKES TRAILING EDGE OF NOZZLE COML
3
IF(1,AE,9) GO TO 40
N6=N6+1
IF(N6-2) GO TO 40
CPYL2=PRP10I*PI1I
CPYL2=CFYL2*(CELY/2.*PI2I2I)
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
GC TC 60
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
CPYL2=0.
GC TC 50
C ACC FORCE AND MOMENT INCREMENTS TO OBTAIN INTEGRATED VALUES
65
CF1=CF1+CFYL2*CFRUI*CFIL2
CF2=CF2+CFYL2*CFYU2*CFYL1*CFYL2
CPV=CPV+CPYL1*CPYU2*CPYL1*CPYL2
CPV=CPV+CPYL1*CPYL1
XLM=X(2I)*X(4I)*X(5I)
TAN9=TAN(CELY/57.29775)
ATAN1=ATAN(CELY/111/57.29775)
GM=1.4
C FIRST TIME THROUGH SUBROUTINE COMPUTE FORCES ACTING ON VEHICLE
IF(1,AE,9) GO TO 233
C UNDERSIDE OF FOREBODY WETTED BY PROPULSIVE STREAM
XLM=X(1I)/COS(CELY)
X12=X(1I)/COS(CELY)
C COMPUTE AVERAGE PROPERTIES FOR SKIN FRICTION COMPUTATION
X12=X(1I)*X(2I)
TAN9=TLAM(2I)*X(1I)*TLAM(3I)*X(2I)/X(1I)
AP12=ID(1I)*X(1I)*X(3I)*X(2I)/X(1I)
PI12=PEES*(LAM(2I)*X(1I)*LAM(3I)*X(2I)/X(1I)
AC112=PI12/(1718.*T)
XPU=2.27E-8*Tool.5/(198.6*T)
V112=(PI2I)*SORT(TMO*TLAM(2I)*X(1I)*M(3I)*SORT(TMO*TLAM(3I)*X(2I)*M(3I)*X(2I)/X(1I)
*12
RE112=MO112*V112*X(1I)*X(2I)/X(1I)
TM=20CO.
RT=1.E6
CALL SKINF(RE112,AM112,TM,T,GM,RT,1,CF)
WRITE(6,342) AM112,TM,T,RE112,RT,CF
342 FORMAT(2X,10HFORE BODY ,1P6E15.7)
C DETERMINE FORCE COEFFICIENTS IN THE BODY AXIS SYSTEM
ANFI=X(1I)*X(1I)*X(2I)-X(3I)*X(3I)*TAN(THETA(1I))-M11/BDI
DF1=AMFI*CF*PM2I2I*LAM(2I)/2.
CFX(1I)=LAM(2I)*TAN(DEL(1I))*CC*DF1*CCSID(EL1)
AMFI2=IV4*X(3I)*TAN(DEL(1I))*X(2I)/BDI
DF2=AMFI2*CF*PM2I2I*LAM(3I)/2.
DZ=X(2I)*LAM(3I)*PM2I2I*GM*CF/2.
CFX(2I)=LAM(3I)*X(2I)*TAN(DEL(1I))*DZ*DF2*CCSID(EL3)
CFX(1I)=LAM(2I)-LAM(2I)*P2I2I*GM*TAN(DEL(1I))*CF/2.-DF1*TAN(DEL1)
CFY(2I)=LAM(3I)*X(2I)-LAM(3I)*M2I2I*X(2I)*TAN(DEL(1I))*GM*CF/2.-DF2*TAN(
DEL(1I))
FC112=CD*0.2
PC112=CFX(1I)*CFX(2I)-FC112
C LEADING EDGE DRAG
C INLET FENCES
CLEF1=CLE/2I
CLEF2=0.01.4*M(1I)*0.2.*SIX(THETA(1I))*0.2.*DELEDI*M3/(BDI*0.1)
C INLET COML WETTED BY PROPULSIVE STREAM
TMO=TLAM(4)
AP3=PI(4)
RE13=PEES*LAM(4I)*X(3I)*C1*M(4I)*SORT(TI)*0.5.2/(1718.*T*XMU)
CALL SKINF(RE13,AM3,TM,T,1,4,RT,3,CF)
FORMAT(2X,10HINLET COML,1P6E15.7)
WRITE(6,343) AM3,TM,T,RE13,RT,CF
CFX(3I)=LAM(4I)*M2I2I*GM*CF*(X(3I)*V4*X(3I)/BDI)/2.
CFY(3I)=LAM(4I)*LAM(4I)*X(3I)
C COMBUSTOR FORCE CALCULATION-VALUE OF CAV45 DETERMINES NEIGHTING
C CF PROPERTIES AT STATIONS (4) AND (5) IN DETERMINING AVERAGE PROPERTIES
C TRANSITION REYNOLDS NUMBER OF RT=1000. WILL GIVE PRECOCINATELY
C TURBULENT VALUE OF SKIN FRICTION
TTO=1.-CAV45*(1.-TEMPS/11)
AP45=PI(4I)*1.-CAV45*(1.-AM5/M(4I))
VC45=V4*(1.-CAV45*(1.-V5/V4))
RT=1.E3
PAV45=PEES*LAM(4I)*1.-CAV45*(1.-P)
XPU=2.27E-8*Tool.5/(198.6*T)
RE4=PAV45*VC45*X(4I)*X(3I)/2.*0.1/(1718.*T*XMU)
GAP45=1.4*1.-CAV45*(1.-GAM5/1.4I)
CALL SKINF(RE4,AM45,TM,T,CAM45,RT,4,CF)
FORMAT(2X,10HCOMBUSTOR ,1P6E15.7)
WRITE(6,344) AM45,TM,T,RE4,RT,CF
CFX(4I)=CF*(PAV45/PDES)*AM45*GM45*(X(4I)*X(4I)*V4/BDI)*X(6I)/2.*X
2I6I)/V4/12.*PI1I)
CFY(4I)=LAM(5I)*X(4I)
RT=1.E5
XLM=X(3I)*X(4I)*X(6I)
TMO=TLAM(6)
AP36=M(6)
XPU=2.27E-8*Tool.5/(198.6*T)

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RE36=PDES*AM(6)*XL36*DI*AM(6)*SORT(TI)*49.2/(11718.*T*XMU)
CALL SKINF(RE36,AM36,TM,T,1.4,RT,5,CF)
CFX(16)=LAM(6)*M(6)*M(6)*M(6)*1.4*CF*XL36/2.
CFY(16)=LAM(6)*X(16)
233 CONTINUE
C CALCULATION OF FORCES ON UPPER SURFACE OF FUSELAGE
RT=1.E6
T=TP*TLAM(8)
APT=PI(8)
XPU=2.27E-8*TOOL*1.5/(198.6*T)
RETP=PDES*AM(8)*XL*DI*AM(8)*SORT(TI)*49.2/(11718.*T*XMU)
CALL SKINF(RETP,AMT,TM,T,1.4,RT,6,CF)
IF(1.EC-9) WRITE(6,345) AMT,TM,T,RETP,RT,CF
345 FCPRAT(2X,10)TOP OF FUS,1P6E15-7)
D3=LAP(8)*M(8)*M(8)*M(8)*XL*GR*CF/2.
CFX(7)=LAM(8)*X*TLAM(DEL7)*D3
PCFP=CFX(7)-C3
CFY(7)=LAM(8)*XL*AM(8)*M(8)*M(8)*X*TLAM(DEL7)*GR*CF/2.
C CALCULATION OF BASE PRESSURE DRAG, BASE PRESSURE IS ZERO
PB=0.
CFX(8)=-PB*VS
C FORCE CN SIDES OF VEHICLE IN WIND AXIS SYSTEM
XA=1.*X(2)*X(4)
B=X4*TLAM(DEL7)*D3
VC12B=.5*VL*VL*X(2)*.5*X(2)*M2*(M3)*X(4)*X(4)*X(5)/2.*.5*X4*X4*T
LANICEL(7)
T=TM
REST=RM*AM(1)*CD*CI*XL/AMU
AMSI=PI(1)
CALL SKINF(REST,AMSI,TM,T,1.4,RT,7,CF)
IF(1.EC-9) WRITE(6,346) AMSI,TM,T,REST,RT,CF
CESE=VC12B*PI(1)*M(11)*GM*CF/BC1
CGFMS=PI(1)*.2*GM*CF*(AMMOZ*AN*11*AMF12)
CESE=CESE*CGFMS
346 FCPRAT(2X,10)FUSE SIDES,1P6E15-7)
C TRANSFER BODY FORCE COEFFICIENTS TO WIND AXIS SYSTEM
CA=CFX(11)*CFX(12)*CFX(13)*CFX(14)*CFX(6)*CFX(7)*CFX(8)
CA=CFY(11)*CFY(12)*CFY(13)*CFY(14)*CFY(6)*CFY(7)
SA=SIN(ALPHAR)
CCSA=CCS(ALPHAR)
VEHICLE NOSE AND EDGE OF INLET COME LEADING EDGE DRAG
CDLEFC=.6*1.4*M(11)*.2*DELD1*.2.
CDB=CA*CCSA*CN*SA*CDSE*COLEBC*COLEF
CLB=CA*SA*CN*CCSA
C REFERENCE NOZZLE FORCE COEFFICIENTS TO SAME QUANTITIES AS BODY FORCES
CFX(5)=CF1*PRATIOFEMIP,GAMMA1)*Y4*LAM(4)*P
CFY(5)=CF2*PRATIOFEMIP,GAMMA1)*Y4*LAM(4)*P
C TRANSFER TO WIND AXIS SYSTEM
TUL=CFY(5)*CCSA*CFX(5)*SA
TUL=CFX(5)*CCSA*CFY(5)*SA
C COMPUTE FLOW PROPERTIES FOR SIDE SURFACE OF WING
IFIPI2-GT,-1.) GO TO 41
T12=TLAM(12)
PI2=LPI(12)
AP122=PI2*12)
LAM(12)=1.
P2(12)=M(11)*M(11)
DEL(12)=DEL(12)/97.29775
IF(DEL(12)-LE-0.) GO TO 19
CALL SMOCK(12,12.88)
GC TC 41
19 CALL PRANT(12)
41 CONTINUE

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C ESTIMATE TRIAL VALUE
DC=INTG/(VTC128*BD1011.1)*.333
C CALCULATE TAKE OFF WEIGHT
16 WTC=INTG128*BC10*DC*DC*ROEST*ATG012*DC*DC/ROEFL1/((11.-PCTP-PCTE
  8/ROEFL*PCTP/ROEFL*PCTE/ROEFL)
FC1=WTC-WTO
C TEST: TC DETERMINE IF CALCULATED IS WITHIN 10 LB OF SPECIFIED TAKE OFF
C WEIGHT
IF(ABS(FD1)-LE-10.) GO TO 17
C DETERMINE NEW TRIAL SCALE FACTOR USING NEWTON-RAPHSON METHOD
FCCT=3.*CDE1*CC*2.42.*CDE2*DC
DC=DC-FC1/FCCT
C CONTINUE ITERATION
GC TC 16
C DETERMINE STRUCTURAL WEIGHT FRACTION
17 STRAC=CDE2*DC*2./WTC
C DETERMINE FUEL WEIGHT FRACTION
PCTVF=1.-STRAC-PCTE-PCTP
C EVALUATE LOG TERM OF BREQUET RANGE EQUATION(STRUCTURAL FACTOR)
SF=ALCG(11.*PCTFCR*PCTVF/(PCTE*PCTP*STRAC))
C COMPUTE CRUISE VELOCITY BASED ON CRUISE MACH AND SPEED OF SOUND AT
C REFERENCE ALTITUDE-W12- CRUISE VELOCITY(FT/SEC) SQUARED
W12=CC*CO*MI1*W11
C COMPUTE RANGE FACTOR OF BREQUET RANGE EQUATION-RF (NAUTICAL MILES)
C VC2=OPTIMAL VELOCITY(FT/SEC) SQUARED
VC2=26000.*26000.
FULPAR=ER*.02*2
R1=V12/((11.-V12/V02)*64.34*FULPAR*6076.)
V8=CLP*CLW*TM1
DEL(8)=(11.428*V6/(W11)*MI1)*CACP)
R1=R1*DEL(8)
C COMPUTE CRUISE RANGE
R=RF*SF
C COMPUTE PRESSURE AT CRUISE ALTITUDE WHERE PCTEA IS PER CENT OF TOTAL
C FUEL USED DURING TAKE OFF, CLIMB AND ACCELERATION
PCTFA=.5*PCTVF
PALT=11.-PCTFA*WTO/(11.-V12/V02)/(V8*DC*DC*BD1)
FCM=FCM+FCUM
R1=R1*CLM1
C COMPUTE SPECIFIC IMPULSE
AISPT=CO*MI1*TM1/((12.17*FULPAR*1.4*MI1)*MI1)*CACP)
RANF=RF*6076.
ALCCT=RFAN*11.-V12/V02/(CO*MI1)*AISPT)
C TEST FOR DECREASE IN CRUISE RANGE IF SO-OPTIMUM NOZZLE LENGTH OBTAINED
IF(CK-EQ-1)WRITE(16,6CD)
FCMAT12X,5HSCALE,3H,4HPALT,3H,3HISP,3H,3HMLD,3H,1HR,3H,2HMF,2H,2H
551,3H,2HMF,3H,2HRS,/)
RANF=RE
SEFCY=SF
IF(LAST.GT.R.AND.DELC(4).LT.5.) RETURN 1
RETURN
492 FCRPAT12X,5HGEOMETRICAL CONSTRAINT AND EDGE OF BING OFF VEHICLE.)
721 FCRPAT13X,1PTE15,7)
334 CONTINUE
WRITE(16,350)
350 FCRPAT12X,4HGEOMETRICAL CONSTRAINT AND THRUST ME DRAG.)
8 RETURN
60 RETURN
END

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818FTC SKIN DECK
SUBROUTINE SKINF(RX,EM,TW,T,GAMF,RT,I,CF)
C THE PURPOSE OF THIS SUBROUTINE IS TO CALCULATE SKIN FRICTION
C AND HEAT TRANSFER COEFFICIENTS
COPPCN/COSF/ ST(10)
DIMENSION G(10)
DATA 67-5,811434E1,3-759829E1,-1-0779506E1,1-121712,-1-78021280E
  9-1,1-1099272E-2,-5-3403789E-4,1-5214431E-5,-2-5064468E-7,1-021E-9/
IF(IST12).GT.(1-1.) GO TO 33
C CALCULATION OF AVERAGE TURBULENT VALUE OF SKIN FRICTION BY SPALDING-CHI
C METHOD(REF26/40)-NCTATION FOLLOWS REF 26
XPU=2.27E-8*OT*1.5/(198.6*TI)
RECCV=.89
A=TW/T
C=RECCV*EM*EM*.GAMF-1.1/(-2.1)
B=1.-C-A
RAD=SCRT(B*B-4.*A*C)
TABW=11.-C)
FC=(ARCSIN((-12.*C*B)/RAD)-ARCSIN(-B/RAD))/SORT(-C))**(-2.1)
FRATAT=(A**(-.702))/(TABW/TW)**.772)
FRATAT/FC
XX=ALCG(FRXXRX)
PCLYR=CG(11)*G(12)*XX*G(3)*XX*2.*G(4)*XX*3.*G(5)*XX*4.*G(6)*XX*5.*
  G(7)*XX*6.*G(8)*XX*7.*G(9)*XX*8.*G(10)*XX*9.
C AVERAGE SKIN FRICTION COEFFICIENT ASSUMING TURBULENT FLOW FOR ENTIRE
C SURFACE
CFTS=(1./FC1)*2.72*POLYB
C CALCULATION OF AVERAGE LAMINAR VALUE OF SKIN FRICTION BY INCOMPRESSIBLE
  LAMINAR ELASTUS SOLUTION MODIFIED BY REFERENCE TEMPERATURE METHOD
TC ACCOUNT FOR COMPRESSIBILITY(REF 15)
C COMPUTE REFERENCE TEMPERATURE
IF(EM-5.12*2.3
  2 TPLT=11.*.032*EM*EM*.58*(TW/T-1.1)
  GC TC 4
  3 TPLT=11.*.023*EM*EM*.58*(TW/T-1.1)
  4 CCONTINUE
  XPUPL=2.27E-8*OTPL*1.5/(198.6*OTPL)
  XRL=441*OTPL*PUL/(TPL*XXMU))
  IF(NT-XX)5,25,25
  5 C COMPUTE SKIN FRICTION COEFFICIENT FOR ENTIRELY LAMINAR BOUNDARY LAYER
  CFL=2.*SCRT(XRL/RX)
  C COMPUTE AVERAGE SKIN FRICTION COEFFICIENT FOR TRANSITIONAL BOUNCARY
  C LAYER BASED ON TRANSITION REYNOLDS NUMBER-RT
  CFTS=CFTS*(11.-RT/RX*(CFL/CFTS))*1.25*(RT/RX)**.625)**.8
  CFCFTAS
  GC TO 30
C USE LAMINAR SKIN FRICTION COEFFICIENT IF REYNOLDS NUMBER IS LESS THAN
C TRANSITION REYNOLDS NUMBER
25 CFL=2.*SORT(XRL/RX)
CF=CFL
30 CONTINUE
ST(11)=CF
RETURN
33 ST(11)=ST(2)
CF=ST(11)
RETURN
END

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SUBROUTINE CCLC(BWAC)
C THIS SUBROUTINE COMPUTES FUEL FLOW REQUIRED TO PROVIDE REGENERATIVE
C CCLCING PER COMBUSTOR AND PORTIONS OF THE INLET AND NOZZLE, ALSO THE
C LEADING EDGES OF THE VEHICLE AND WINGS

      CCMCA /VAR/M(13),DEL(13),DEL(13),LAM(13),TMT(13),MCWL,M2(13),
      IR(1),YA,Y5,CPE(13),CFV(13),CEL9,ALPMA,CACAP,TLAP(13),FACSS,
      ZER,APS,P
      CCMCA /NAMEC/ TEMPS,V4,V5,MINS,CAMS,BD1,CAV45,AS,AR4,TEPP65
      CCMCA /VAR2/ P12,PASPR,MF,PDES,PAL,T,MTOO,MTOY,CUT,PALIT,CPALT,FSD
      CCMCA /ACCL/ OLE,O1,MTO
      CCMCA /COSZ/ ST101
      REAL P,M2,LAM
      C CCONSTANTS USED IN SUBROUTINE
      CAP=1.0
      RACT=9719.
      C CM2-SPECIFIC HEAT CAPACITY OF LIQUID HYDROGEN(18TU/LB)
      CP=24000.
      C GJ2-USEC IN CONVERSION OF UNITS OF ENTHALPY FROM MECHANICAL TO THERMAL
      GJ2=32.1778E-02.
      C FST-STOCHIMETRIC FUEL-AIR RATIO- HYDROGEN-AIR
      FST=.0292
      C PR-RECOVERY FACTOR USED IN DETERMINATION OF ADIABATIC WALL ENTHALPY
      PR=.89
      R=RACT/28.9
      C DETERMINE AMBIENT TEMPERATURE AT REFERENCE ALTITUDE
      CALL ATMOS(MF,TM,SIGMA,RMO,TMETT,DELTA,CO,AMU,K)
      C2=1.0/(CM2*FST*2.1)
      RLE=CLE/2.
      GSTAG=.729*885.0*(M1+CO/1.64)*.73*(1-SIGMA/RLE)*.5
      CLE=RT*1012.4*MBU*1.02*(M1+CO/1.64)*.5
      ECLC=CLE/(1+STAG*CO*MBU)*CACAP*0.1*0.6D1*CMH2*32.2)
      WRITE(6,45) GSTAG,OLE,ERCLE
      45=1000
      C CCMPT- COOLING REQUIREMENT FOR UNDER SURFACE OF FUSELAGE OPPOSITE
      C INLET COWL
      C DETERMINE RATIO OF COOLED AREA TO PROPULSIVE STREAM TUBE AREA IN PLANE
      C NORMAL TO COOLED SURFACE
      C AIAC3=RT/((COS(DEL1))*(1+YA*(13)+TAN(DEL13)))
      C DETERMINE VELOCITY IN PROPULSIVE STREAM TUBE
      T3=TLAP(13)*TM
      V3=M1*330*SORTIGAM*0.3*TS1
      V32=V3*V3
      C DETERMINE STATIC ENTHALPY IN STREAM TUBE(SUBROUTINE AIRTAB TABLES BASED
      C ON DATA FROM REF 29 AT ZERO EQUIVALENCE RATIO)
      CALL AIRTAB(73,F3)
      MAMB3=M3*PR*V32/GJ2
      T3=2000.
      C WALL ENTHALPY
      CALL AIRTAB(73,M3J)
      MAMB3
      C3=ST(13)/C2
      C COOLING EQUIVALENCE RATIO FOR SURFACE
      FENC=C3*AIAC3*(MAMB3-MW)/COS(DEL13)
      IF(FENC-LT-.0) FERQ1=0.
      C CCMPT- COOLING REQUIREMENT FOR INLET FENCE ALONG SHOCK FROM
      C SECCO INLET RAMP
      BRAC3=X(12)/1001*(COS(DEL13))
      FENC=C3*BRAC3*(MAMB3-MW)
      C CCMPT- COOLING REQUIREMENTS FOR INLET COWL AND SIDES
      ALERCA=X(13)*(1.01*.04/BD11)/YA

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TEPP4=TLAM(4)*TM
CALL AIRTAB(74,M4)
MAMB4=M4*PR*V4*V4/GJ2
C3=ST(13)/C2
C COOLING REQUIREMENT FOR SURFACE
FENC=C3*ALERCA*(MAMB4-MW)
C CCMPT- COOLING REQUIREMENTS FOR COMBUSTOR
ASCAC4=2.0*X(14)*(1.1/Y4*0.1/BD11)
MAMB5=M1*F5*PR*V5*V5/GJ2
C AVERAGE ADIABATIC WALL ENTHALPY BETWEEN ENTRANCE AND EXIT OF COMBUSTOR
MAMB5=MAMB5*(1.1-CAV45*(1.1-MAMB5/MAMB4))
C3=ST(14)/C2
C COOLING REQUIRE FOR COMBUSTOR
FENC=C3*ASCAC4*(MAMB5-MW)
C CCLCING REQUIRED FOR NOZZLE COWL AND SIDES
ATEAC5=X(16)*(1.1+YA/BD11)/YA
C COOLING REQUIRE
FERCT=C3*ATEAC5*(MAMB5-MW)
C CCLCING REQUIRED FOR AFT-FUSELAGE SURFACE OPPOSITE NOZZLE COWL
ATEAC4=X(16)/(COS(DEL5))*(1+YA*(16)+TAN(DEL5))
TEPP6=TEMP5+TEPP65
V6=M6*330*SORTIGAM5*MS*TEMP6)
MAMB6=MAMB5
C3=ST(10)/C2
FERC=C3*(MAMB6-MW)*X(6)/(1001*(COS(DEL5))
C TOTAL CCLCING REQUIREMENT FOR SURFACES CONSIDERED
TFERC=FFERQ1*FERCLE+FERQCA*FERCTE+FERQNZ*FERCF1+FERQNF*ERCLE
WRITE(6,700) FERQ1,FERQLE,FERQCA,FERCTE,FERQNZ,TFERC
700 FCMAT(124,36)*PHI*5 FROM INLET TO NOZZLE AND TOTAL,/.2X,6F13.6//)
      RETURN
      END
C *****END OF PROGRAM LISTING*****

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| 13. ABSTRACT An optimization of air breathing hypersonic cruise vehicles was performed in order to determine basic configuration characteristics and performance trends. A distinctive feature of the investigation is that prediction techniques such as the method of characteristics were used to determine the flow field surrounding the vehicle; therefore, any interaction between the aerodynamic and propulsive flow fields is accounted for in a fundamental manner. The general class of vehicles considered in the study cruise in the Mach 8-12 speed regime, utilize a hydrogen fueled supersonic combustion ramjet engine, and can be geometrically characterized as two-dimensional wedgelike shapes. Configurations were optimized for maximum cruise range as determined from the Breguet range equation which incorporates a measure of the aerodynamic, propulsive, and volumetric efficiencies of a configuration. A generalized configuration model was defined by a discrete parameters which transformed the variational problem to a static or discrete optimization problem. The direct method of function optimization, utilizing search algorithms such as random point and adaptive creeper techniques, was employed to determine the value of the parameters defining the optimum configuration for cruise at design Mach numbers of 8, 10, and 12. The design parameter space in the vicinity of the optimum point was explored to show performance sensitivity. Results of this study showed that optimum cruise configurations are characterized by small wings. Approximately three-fourths of the total lift is provided directly from the propulsive system in accordance with an expression for optimum propulsive force vectoring derived in the study. The results | | |

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also indicated that Breguet range factors of approximately 10,000 nautical miles are attainable by vehicles which cruise in the Mach 8 to 12 speed regime.

| 14 | KEY WORDS | LINK A | | LINK B | | LINK C | |
|----|---|--------|----|--------|----|--------|----|
| | | ROLE | WT | ROLE | WT | ROLE | WT |
| | Configuration optimization Hypersonic vehicles Computer-aided design Direct search | | | | | | |